U. S. ARMY
TRANSPORTATION RESEARCH COMMAND
FORT EUSTIS, VIRGINIA

TRECOM TECHNICAL REPORT 63-18

AN INVESTIGATION OF MECHANICAL STABILITY AND TRIM AUGMENTATION ON HELICOPTER IFR CAPABILITY

Task 1D121401A14174
(Formerly Task 9R38-01-017-74)

Contract DA 44-177-TC-791

June 1963

prepared by:

CESSNA AIRCRAFT COMPANY
Military Division
Wichita, Kansas

HEADQUARTERS
U S ARMY TRANSPORTATION RESEARCH COMMAND
Fort Eustis, Virginia

This report was prepared by the Military Division of Cessna Aircraft Company under the terms of Contract DA 44-177-TC-791. Views expressed in the report have not been reviewed or approved by the Department of the Army; however, conclusions contained therein are concurred in by this Command.

Electronic stabilization devices have had widespread usage in nearly all models of rotary-wing aircraft. This "black box" concept, while effective, is a complex and costly system. In an effort to reduce complexity and provide for ease of maintenance, Cessna Aircraft Company designed and flight tested a mechanical stabilization system utilizing a YH-41 helicopter. The system is applicable to all helicopters and is discussed in detail in this report.

FOR THE COMMANDER:

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Prepared by:

CESSNA AIRCRAFT COMPANY
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SUMMARY

A development and test program has been conducted by the Cessna Aircraft Company under contract to the U. S. Army. This program has been directed toward the development of improved helicopter instrument flight characteristics. Modification of the basic aircraft and the addition of stabilizing systems have been accomplished and tested. All modification and additional systems are of the mechanical type. Electronic stabilization devices were not considered during this program.

Directional flight characteristics were improved with tail rotor blade design changes and with the addition of two mechanical gyro systems. Evaluation of these systems was accomplished by measurement of the aircraft characteristics and by pilot evaluation during simulated instrument flight.

The mechanical stabilization systems were found to be effective and to offer a low level of required maintenance. The IFR capability was extended to 25 knots on a basis comparable with 40 knots as determined on a previous program. Simulated IFR approaches, transition to hovering, and landings were conducted.

The investigation was sponsored by the United States Army Transportation Research Command.

CONCLUSIONS

In accordance with the results of the test program the following principal conclusions were made.

1. Improved directional flight characteristics can be obtained by proper design of the tail rotor and tail rotor control linkage. Looseness and flexibility in the control system can distract from the directional characteristics and may reduce apparent directional damping and introduce a residual directional oscillation. Proper placement of the feathering axis, and chordwise center of gravity with respect to the center of pressure on the blade can result in improved directional characteristics of the helicopter.

2. The problem of holding heading becomes more difficult as the flight speed is reduced due to the increased influence of small roll displacement errors upon the rate of turn due to these displacements. A mechanical gyro which senses yaw rate and corrects for this rate by a small lateral control change has been tested and shown to be a benefit with regard to reducing the heading drift.

3. Damping about the yaw axis has proved to be quite beneficial. This damping input may be limited in authority to a small value and still be quite effective. With this limited authority, loss of controllability is not significant. During this program a mechanical yaw damping system, employing a mechanical gyro operating an unboosted control system proved to be very beneficial.

4. It is concluded that MIL-H-8501A; 3.6 (instrument flight characteristics) should include requirements with regard to heading hold capability with the controls unattended. Although the scope of the test program has not been of sufficient extent to establish specification values it appears that a capability of holding ±5 degrees heading for a period of one minute without pilot attention, and for those flight conditions employed during IFR flight, is sufficient.

INTRODUCTION

Applied research concerning the handling characteristics of helicopters has been conducted during the past five years. This research has revealed benefits from design considerations on the basic aircraft configuration, and stabilizing systems have been developed which improve these characteristics.

Throughout these developments only mechanical and aerodynamic configurations have been employed. Electronic stabilization has not been introduced into the programs. These mechanical systems have proved to have several advantages and some disadvantages in comparison with the electronic approach. The mechanical system is designed, manufactured, and maintained on the basis of the same principles as the aircraft. With the mechanical systems, the level of reliability attainable and maintainable can be the same as the basic aircraft structure and mechanisms. The mechanical systems do not introduce a different philosophy of maintenance requirement. Personnel trained to maintain the basic aircraft find these systems to be easily understood and the maintenance directly in line with their experience. In other words, a "black box expert" need not be added to the maintenance crew.

The mechanical gyro system does not accomodate the wide variety of functions offered with electronic type autopilots. Functions including heading or navigational course locks, and controlled turn rates were not accomplished with the mechanical systems.

These mechanical systems have proved successful in augmenting the handling characteristics but do not offer the wide range of functions offered by the electronic systems. The ability to "lock" on a heading, automatically programs a flight sequence, or follow a flight path established by ground facilities, this has not been accomplished with the mechanical systems.

The applied research concerning handling characteristics was initiated by the Cessna Aircraft Company. The developments of this initial program, including a mechanical gyro for roll stability augmentation, were introduced into the commercial CH-1C model on a production basis. This work was continued on contract AF33(600)-2857 in cooperation with the U. S. Army, resulting in the development and FAA approval of the CH-1C (IFR). The completion of this program provided the first helicopter to be FAA approved for operation under instrument flight rules, and assisted in developing flight standards for IFR approval.

The FAA approved flight manual, for the CH-1C (IFR), permits instrument flight throughout the speed range from 45 knots to V_{NE}. The 45 knot minimum speed limitation was established as a result of stability deterioration at the lower airspeeds. Accordingly it is recommended that instrument flight be conducted at 60 knots to V_{NE} whenever possible. It appears that the additional systems investigated during this program would allow the lower speed limitation to be placed at 25 knots. The test aircraft, however, was not presented to FAA for their approval. On two occasions, simulated GCA approaches and landings were completed. However, routine operation including the landing will require a pilot presentation more complete than employed on this program.

U. S. ARMY PILOT EVALUATION

YH-41 USA A/N 56-4236 was flown a total of 5.33 hours in five flights during 12 and 13 December 1962, by two Army aviators of USATRECOM. These flights were made to evaluate the stabilization research efforts performed under Contract DA44-177-TC-791.

The helicopter was flown in various configurations of stability augmentation, both under visual and simulated instrument conditions.

The following general comments are offered:

The system, as installed, is a simple, reliable mechanical system.

With all axes stabilized, the helicopter is simpler to fly in all speed regimes. The stability about the pitch and yaw axes deteriorates with decreases in speed, but it appears that hooded flight at 25 KIAS is possible with some difficulty.

Stability about the pitch and roll axes is considered the most important from a safety aspect. At 40 KIAS, the pitch axis demonstrates a poorly damped phugoid which makes precise airspeed control difficult. At higher speed, the phugoid is more heavily damped and is satisfactory.

The roll stability is good at all speeds investigated and considered desirable.

The directional stability augmentation gyros, both yaw damper and heading assist, were desirable. With yaw augmentation inoperative, the hooded tasks were considerably more difficult and yaw excurions even in relatively still air were difficult or impossible for the pilots to control more closely than ± 10 degrees of desired heading.

With the yaw augmentation system operative, a long-period heading drift remains. This is undesirable and complicates the pilot's task. It appears that a heading lock feature would overcome this mild deficiency.

These stability augmentation systems, which are applicable to any helicopter, are desirable and are particularly so in application to an instrument trainer or to any helicopter to be used under actual weather conditions.

CESSNA PILOT EVALUATION

Simulated GCA flight in the CH-1C helicopter is made much easier by the use of the yaw damping gyro and the heading assist gyro. The effect of these systems is even more noticeable in turbulent air than in stable air. This results from the consideration that the contribution offered by the systems is more noticeable when the magnitude of the atmospheric disturbances causes yaw rates in excess of the threshold of sensitivity of the systems.

The yaw damping gyro appears to be the most beneficial of the two systems for instrument work. Without this system, random yaw oscillations due to gusts are present which become very disconcerting. With this system activated, yaw oscillations are cut down to a point where the pilot is not aware of yaw disturbances.

The heading control gyro, although a definite aid as a heading lock, falls into a category of secondary importance when compared to the yaw damping gyro as an aid to instrument flight. This does not imply that the heading assist gyro is not doing a job. It is definitely a help in maintaining a heading. Of the two systems, the one that is missed the most is the yaw damping gyro when it is deactivated.

GCA approaches, down to 25 knots airspeed, are possible at sink speeds up to 750 feet per minute. The low-speed approaches are more difficult to make than faster approaches. This is probably due to the fact that pitch stability deteriorates at the slower speeds and also that turning rates increase rapidly at small bank angles. Approaches in autorotation at 50 knots present no problem.

Various tail rotor configurations were evaluated to determine which one offered the most damping. The original tail rotor, Type A, provided the least yaw damping. When this tail rotor was redesigned into Type B, it gave a much more stable ship about the yaw axis. Type C tail rotor was designed next which provided damping that was comparable to Type A tail rotor. Type D tail rotor was later developed which has damping characteristics that are as good or better than Type B tail rotor. This last design tail rotor is the one that was used the most during the stability program.

With the straight-thru hub, the result of sweeping the blade is to put the center of pressure behind the feathering axis, which has a stabilizing effect.

With the coning tail rotor, the stabilizing factor here is to have the center of gravity and the center of pressure coincide and to be located slightly behind the feathering axis.

COMMENTS CONCERNING MIL-H-8501A, "HELICOPTER HANDLING QUALITIES; GENERAL SPECIFICATION FOR"

The development and testing accomplished on this program were directed toward the improvement of helicopter directional characteristics. At low airspeeds, the heading control imposes a problem even with an aircraft possessing a high degree of static and dynamic stability. The testing accomplished on this program indicates that the stability requirement of MIL-H-8501A, 3.6.1.1, is satisfactory from a damping standpoint. It appears, however, that an additional requirement should be imposed which will define the aircraft's ability to maintain a heading in an atmosphere of moderate disturbances.

At low airspeeds, an aerodynamic reference is not sufficient to provide for heading control. The testing on this program indicates that a space reference is necessary in order to maintain the desired heading. The requirements of 3.6.1.1 will be met, in many cases, with stability augmentation. With stability augmentation, the heading hold capability can be accomplished and would offer great assistance during low-speed blind flight.

The scope of the test program has not been of sufficient extent to establish the degree of heading hold capability for specification purposes. A capability of holding ±5 degrees heading for a period of one minute without pilot attention and for those flight conditions employed during IFR flight would seem to be sufficient.

HEADING ASSIST GYRO

During simulated and actual instrument flight in helicopters it has been noted by several observers that the heading control becomes more difficult as the airspeed is reduced. To enjoy the full potential of the helicopter, it is essential that the blind operations be satisfactory at reduced speeds. In consideration of the lateral-directional characteristics of a stable aircraft, it becomes apparent that the heading deviation rate resulting from the roll displacement increases as the speed is reduced in the inverse ratio to the speed. The pilot, flying under the hood, thus finds that an error in roll attitude causes an excessive turn rate at low airspeeds. At reduced airspeeds, the attitude gyro display does not provide sufficient accuracy to prevent a slight roll attitude error, which results in a significant heading change rate. The following table shows the heading deviation resulting from a 2 degree bank angle after 10 seconds.

Airspeed (Knots)	Heading Deviation After 10 Seconds (Degrees)
20	19.1
40	9.5
60	6.4
80	4.8
100	3.8
150	2.5

The heading assist gyro installation was designed and tested to assist the pilot with the heading drift problem. A schematic drawing of the system is shown in Figure 4. The installed system is presented in Figure 5. Measured flight test performance is shown in Figure 6.

The system employs a damped rate gyro oriented to sense yaw rate. Precession of the gyro introduces a lateral cyclic input which rolls the helicopter "out of the turn". The cyclic input is introduced through a mixing linkage with the pilots controls and allows either the gyro or the pilot to introduce lateral control displacements independently and simultaneously. Assuming that a roll displacement deviation, from level attitude, is present, the helicopter will assume a yaw rate in a turn. The yaw rate will actuate the gyro, which will introduce a lateral cyclic control correction and will in effect remove the original roll displacement error.

The system provided substantial aid to the pilot by reducing the heading drift and allowing a longer instrument panel scanning rate. Figure 6 presents the typical results obtained from the device. These data were recorded with the aircraft in flight and unattended by the pilot for the time period shown. It will be noted that the heading drift after 60 seconds deviated 45 degrees at 80 knots without the gyro activated. With the gyro activated, this deviation was reduced to 4 degrees. At 40 knots, the deviation was 60 degrees with the gyro deactivated and 20 degrees with the gyro active. The inverse effect of speed and the beneficial effect of the gyro on the heading drift rate are apparent from these data.

Several factors were considered in the design of the heading assist gyro. In translational flight, the lateral and directional characteristics of the helicopter are coupled. In hovering flight, it is desirable to have the lateral and directional characteristics uncoupled. In order to avoid a coupling problem in hovering flight, the authority of the gyro was limited to a very small value, 2.5% of the full cyclic range. With this small amount of authority, the control coupling effects in hovering flight were imperceptible and yet adequate authority was available to assist the pilot materially in translational flight.

To accomplish the desired effect, the device must sense small yaw rates. With the mechanical gyro of this type, a threshold of sensitivity exists due to the control linkage friction. The employment of low authority was compatible with a low threshold of sensitivity, since both could be accomplished with a high mechanical advantage between the gyro and the control linkage.

The introduction of the gyro system provides the helicopter control system with another degree of freedom. In order to avoid the natural frequency of this additional system from appearing in the motion of the aircraft, the gyro system must be critically damped. A viscous damper was installed to accomplish this damping. The viscous damping applied to the gyro motion, in effect, integrates the gyro rate signal, thus producing a displacement component to the signal introduced into the cyclic control. This, in effect, gives the heading assist gyro a "memory".

The gyro centering spring was designed to be easily adjustable in flight. The proper magnitudes for the viscous damper and the centering spring were determined by flight test evaluation.

Power controls are considered essential for the successful application of this system. A manual control system which has normal friction loads or control load variations would require a heavy gyro system. Power controls, which eliminate all control forces and have a friction load in the area of .5 pound, make a heading control system weight of 4 to 8 pounds practical.

The system has been found suitable to accomplish the intended pupose. Both the severity and rate of heading errors have been reduced to a level which relieves the pilot from intensive heading control during IFR approaches. The severity of the heading control problem has been reduced because this system is spiral stabilizing. The unmodified aircraft would roll off into a steep spiral if a roll disturbance was allowed to go unchecked. The aircraft with the heading control system operative would not enter a spiral, and thus eliminated a catastrophic maneuver. The heading control system reduced the rate of heading error development to a level which allowed reduction of the pilot scanning rate. However, the device is a function of yaw rate without an absolute reference and is subject to accumulative heading errors. The device will not, then, provide ultralong-period heading control for cross-country flying as will an autopilot with the heading or course lock feature. The characteristics of the heading assist gyro are presented on the data sheet, Figure 7.

TAIL ROTOR DESIGN CONSIDERATION

Helicopter directional dynamic stability has been one of the primary areas of investigations during the performance of this contract. During the previous stability program under Contract AF33(600) 37864, considerable improvement in dynamic directional stability was made by altering the tail rotor design. It was felt that additional improvement could be made and was thus investigated during this program.

The first configuration tested (Configuration A) consisted of a two-blade semirigid tail rotor (see-saw). The blades were retained in a single hub at a fixed precone angle. The complete tail rotor assembly was free to flap (see-saw) on the flapping hinge. Reference Figure 8. The blades were manufactured as a single wrap-around aluminum sheet and formed a blade with planform and thickness taper. The feathering axis of the blade was located behind the center of pressure, and the blades and hub assembly formed a rigid beam against flapwise bending. With this relationship between the center of pressure and the blade feathering axis, the blade pitch change moments resulting from tail rotor inflow are destabilizing.

The second tail rotor configuration tested (Configuration B) was identical to Configuration A except for the rotor blade. On Configuration B, the wrap-around skin on the blade was formed to provide a rearward sweep to the blade. Reference Figure 8. This change placed the center of pressure of the blade, at the 3/4 radius station, behind the feathering axis. Flight testing of this configuration demonstrated a substantial improvement in directional characteristics.

In consideration of the structural benefits which are forthcoming from the use of individually hinged blades, two tail rotors of this type were tested. Configuration C employed a rotor hub with individually hinged blades and with the center of pressure ahead of the center of gravity. Reference Figure 9. The blade planform corresponded to Configuration B. The center of gravity of this blade was forward from the location on blade B. Flight testing of Configuration C indicated a deterioration in directional stability due to the introduction of blade coning freedom.

Configuration D was identical to Configuration C except for the blade center-of-gravity location. Reference Figure 9. By redesign of the blade skin and the addition of a stainless-steel leading-edge cap, the chordwise center-of-gravity location of the blade was moved forward to the 25% chord location. A substantial improvement in directional stability was experienced.

From the experience gained on this program, it is apparent that the detail design of the tail rotor can have a strong influence upon the directional dynamic stability of the helicopter. The degree of control system flexibility and any control system looseness are important factors in determining the directional characteristics of the aircraft. Although a tail rotor pitch change mechanism may be redesigned to be irreversible at a point close to the rotor, some degree of flexibility and looseness will be present after service wear has accumulated. With this control system freedom, the blades will change pitch in accordance with the blade feathering moments applied. If these moments change the pitch in a manner to cause a deterioration of the helicopter stability, this can be expected to occur with accumulation of service time. If, however, the tail rotor design is arranged to provide stabilizing feathering moments, then blade freedom will not destabilize the helicopter.

Tail rotors A, B, C, and D were flight tested on a Cessna YH-41 helicopter. The helicopter was equipped with a sideslip angle indicator; the results were recorded by an oscillograph. The pedal controls were provided with an adjustable stop mechanism which allowed a predetermined amount of control movement to be introduced. A pedal pulse input was employed for the evaluation. The method of test consisted of the following:

a. Trim helicopter for straight and level flight in smooth air at approximately 3000 feet density altitude.

b. Pulse the pedal control.

c. Record the sideslip angle with respect to time until the oscillation ceases.

d. Conduct testing at 40 and 80 knots and with right and left pulses for each tail rotor.

The results of these tests are shown on Figures 14, 15, 16, and 17. Tail rotor A is shown to be the least damped for the four rotors tested. This tail rotor also had a very high and erratic residual sideslip oscillation which was quite troublesome to the pilot. During past development programs, this tail rotor has actually developed divergent oscillations during pedal free flight. Tail rotor B, which represents the stable version of the see-saw rotor, showed a marked improvement in damping as well as elimination of the residual oscillation. Pilot opinion rated tail rotor B far superior to tail rotor A. Tail rotor C, representing the unstable coning tail rotor, provided more damping at both 40 and 80 knots than tail rotor A, but less damping than rotor B at 80 knots. Tail rotor C had a residual sideslip oscillation which was somewhat worrisome to the pilot but too small to appear on Figure 16. The residual magnitude was approximately ± 1 degree. This tail rotor did not have any of the erratic characteristics demonstrated by tail rotor A. Tail rotor D, representing the neutrally stable coning rotor, had improved damping over rotor C at 80 knots. No residual oscillation was apparent from rotor D. Residual oscillations had a strong effect on pilot opinion; consequently, rotor D received a better rating than rotor C even though the difference in damping was small.

In consideration of applying the findings of this investigation to tail rotor system designs, the following design considerations are important. First, the tail rotor control system rigidity and tightness of all linkage are important.

On semirigid tail rotor configurations wherein the lift loads on the blades are carried along the blade to the hub by means of beam bending and shear, the primary consideration is the relationship between the center of pressure and the feathering axis. Best results can be expected if the center of pressure and feathering axis are coincident or if the center of lift is behind the feathering axis. With the center of lift ahead of the feathering axis, residual directional oscillation can be expected.

On a tail rotor employing free flapping and coning blades, the relationship between the blade chordwise center-of-gravity location and the chordwise center-of-pressure location is of primary interest. This is true since the lift loads are balanced primarily by centrifugal force components perpendicular to the blade and resulting from blade coning. The best flight characteristics can be expected with a fully mass balanced blade. If the center of gravity is behind the center of pressure, residual oscillations in yaw and reduced directional damping may result.

Figure 18 presents a summary of the tail rotor blade characteristics for the tail rotors tested.

DIRECTIONAL DAMPING GYRO

In consideration of improving the directional flight characteristics, the addition of damping of helicopter motion about the yaw axis was investigated. In accordance with the philosophy of this program, this damping was introduced with a mechanical system. The test helicopter employs an unboosted directional control system. Control motion is transmitted from the pedals by means of cables to the rear of the lower tail boom and by push-pull control to the tail rotor. The linkage is reversible throughout.

Satisfactory mechanical gyro systems have been employed on this program and prior to this program wherein the mechanical gyro operated through a boosted system. This mechanical gyro, however, represents the first installation by this contractor of a mechanical gyro stabilizing system installed on unboosted control.

The directional damping gyro was installed in the tail boom. Reference Figures 20 and 21. The design was arranged so that the gyro input motion into the directional control was introduced through a mixing linkage. Reference Figure 19. In this manner, the input motion from the gyro did not move the pedals. On previous installations operating in boosted control systems, a viscous damper was necessary to damp the gyro motion. On this installation, however, the damper was found to be unnecessary. The control system friction was found to provide sufficient damping.

In order to present satisfactory flight characteristics during all flight conditions, the gyro authority was limited to a total of 15% of the total available directional control. Thus, in gusty air the gyro can introduce a tail rotor pitch change equal to ±7 1/2% of the total pitch range to effect damping of the aircraft. This introduces a yaw velocity limit above which the gyro does not introduce additional input. With this system, directional damping of the aircraft is sufficient to maintain steady flight in turbulent air, and this is accomplished without a noticeable effect on maneuverability. Figure 22 presents a summary of the directional damping gyro characteristics.

The system was evaluated by the application of pedal pulses in smooth air, level flight. Reference Figures 25 and 26. Records were also taken with unattended flight in turbulent air with the gyro active and inactive. Reference Figures 23 and 24. In level flight, the maximum gyro travel was obtained when the pedal pulse produced a sideslip angle of 9 degrees at 80 knots and 13 degrees at 40 knots. A pilot pedal position change of .38 inch would bottom the gyro and allow full control effectiveness beyond this limit. Thus, hovering turns could be conducted without a noticeable loss of control power. The gyro would, of course, be bottomed against the stop for the complete maneuver.

In evaluating the mechanical gyro stabilizing system, comparison to a single axis autopilot is essential. As previously discussed, there exists an obvious cost and maintainability advantage and a probable weight and reliability advantage. However, the autopilot offers flexibility not available with the mechanical approach. Autopilots may readily be equipped with heading or navigational course locks. Pilot-selected turn rates controlled by the autopilot are also available.

From the foregoing discussion, two conclusions may be drawn:

1. The mechanical gyro stabilization system is a simple system which provides good yaw damping.

2. The mechanical stabilization system has the capability to be cheaper, lighter, more reliable and more maintainable than an autopilot; but the mechanical system cannot provide the more sophisticated features of an autopilot, such as heading, course lock, and controlled turn rate.

AUTOMATIC DIRECTIONAL TRIM

Coupling between flight controls adds to the pilot's task of flying the aircraft. The single-rotor helicopter has a characteristic directional trim change associated with power changes. This characteristic has been universal on this type of aircraft and has accordingly been accepted. Nevertheless, the requirement for pedal correction with each power change burdens the pilot with one more task, the elimination of which would offer assistance to the pilot, especially during blind flight when comprehension of the status of the flight situation is more difficult.

Because of the high tail rotor location of the Cessna helicopter, a similar coupling between roll and power was present on early models. This coupling, being somewhat unique to this aircraft, was not readily accepted by helicopter pilots. The coupling was successfully eliminated by the addition of a power trim cylinder which automatically accomplished the trim change, thus relieving the pilot of this task. The engine installed in this aircraft employs a gear-driven supercharger. The pressure rise across this supercharger is related directly to the engine power for all power conditions. With this known relationship, a simple device was incorporated which effectively and reliably accomplished the desired lateral trim change. The device consisted of a spring-loaded piston in a closed cylinder. The ends of the cylinder were connected to the induction system so that the supercharger pressure opposed the spring load. With this configuration, each power setting produced a determined position of the piston in the cylinder. The motion of the piston is transmitted to the lateral control system through suitable mixing linkage, thereby accomplishing the desired trim change.

The use of the automatic trim cylinder on the lateral system has proved to be very satisfactory from a performance and reliability standpoint. In accordance with this excellent experience with this device, it was decided to install a unit on the directional axis to provide automatic trim correction with power changes on this axis.

A mixing linkage was designed to incorporate the trim device into the tail rotor control system. The mixing of the controls was accomplished at the firewall position. The same trim cylinder, piston, and spring combination was employed. The installation of the trim cylinders was in the lower engine compartment. Reference Figure 27.

The performance of the system was not considered satisfactory. The lateral system operated with control boost assistance; accordingly, the forces supplied by the trim cylinder were sufficient for precise control. In the case of the directional system, the control friction and tail rotor forces were excessive for the unit tested. Redesign of the unit to provide higher power would undoubtedly provide satisfactory operation. This redesign was not, however, accomplished during the subject program.

TEST EQUIPMENT

1	Recording Oscillograph Type 5-116 Pe-14 SN666BA3 Consolidated Equipment Corp.
4	Control Position Transducers Cessna Ft 40-28
	Doel Cam Rate System
1	Demodulator Power Supply Type VFDD-1 Minneapolis - Honeywell
1	Filter Minneapolis - Honeywell
1	Dynamotor Type D-3 Generators Inc.
1	Inverter X3499-1 Eicor
3	Rate Gyros Model K-3 - Pitch, Model K-4 - Roll, Model K-56 - Yaw Minneapolis - Honeywell
1	Bridge Box Cessna Property
1	Attitude Gyro Type K-3 Minneapolis - Honeywell
1	Accelerometer AJ17A-3-120 Statham Laboratories
1	Battery Voltabloc Type 10-VO-9 Saft Corp. of America
	Yaps Head Cessna 20724

COMMUNICATION AND NAVIGATION EQUIPMENT

VHF Communications Type 210

ARC Type	RT-11A Transceiver
	P-15A Power Unit
	C-67A Control Unit
	A-15 Antenna
ADF Type	21A (ARC)
	L-11 Loop
	R-30A Receiver
	IN-12 Single Indicator
	C-59A Control Unit
	P-14A Power Unit
	20557 Sense Antenna Installation
VOR/Localizer System (ARC)	
	R-34A Receiver
	B-13A-1 Converter
	IN-10 Course Indicator
	C-88A Control Unit
	A-21 Antenna
Glide Slope (ARC)	
	R-31A Glide Slope
	Glide Slope Antenna Type 37P-3 (Collins)
R20 Marker Beacon (ARC)	
	Antenna -541 7409 003 (Collins)

F-14A Interphone Amplifier

2 MX1646/AIC Head Set Adapters
2 Head Sets H-79 A/IC (Roanwell Corp.)
2 Microphone M-33/AIC (Roanwell Corp.)

FLIGHT ATTITUDE EQUIPMENT

Lear	4005 Attitude Indicator System
	4005H Indicator
	7000E Gyro - Vertical
	2159P Rate Gyro
	AN 3499-1 Inverter

Sherry Type C1 Directional Gyro

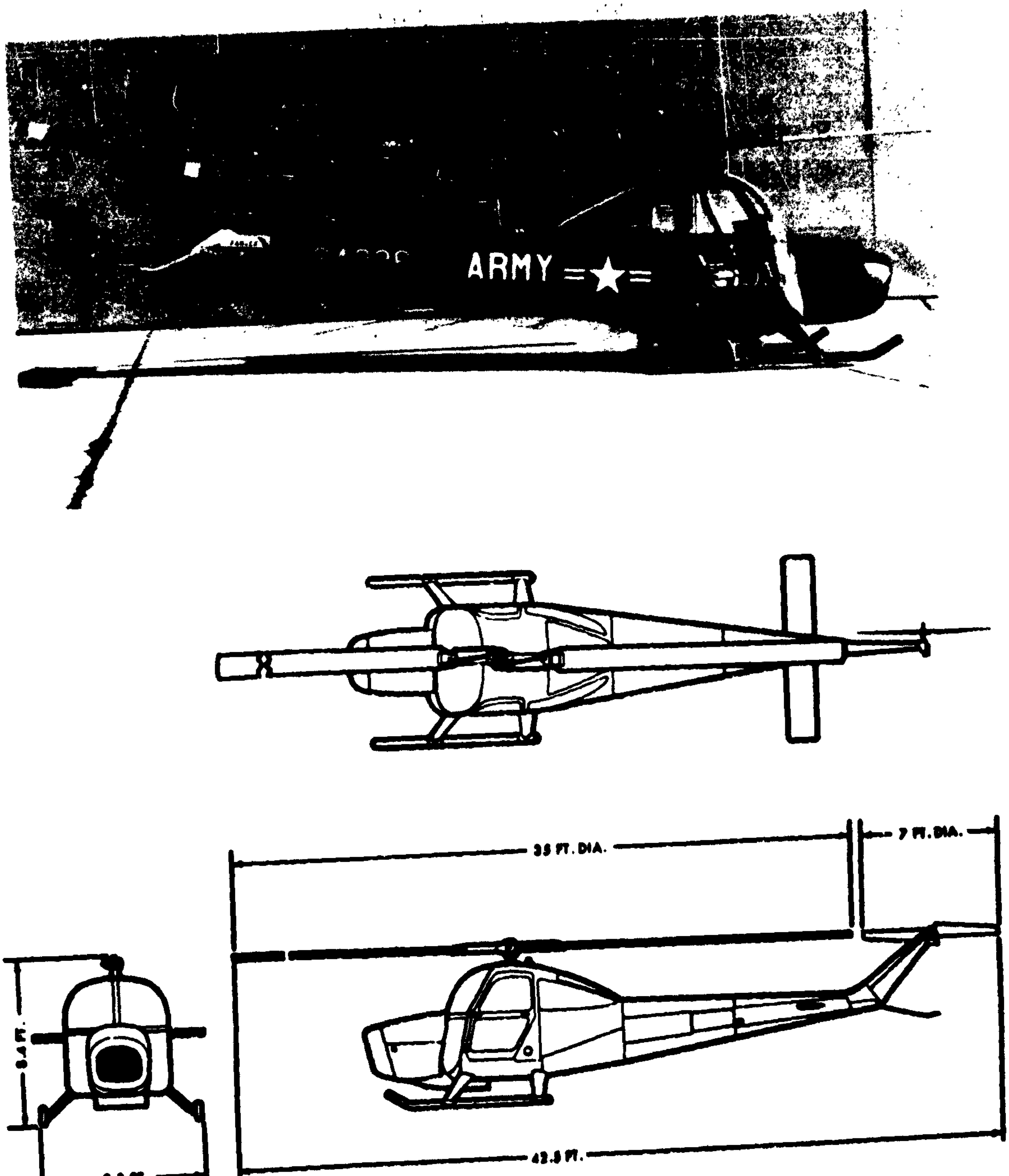


FIGURE 1. TEST HELICOPTER, U. S. ARMY YH-41 (MODIFIED), S/N 56-4236.

FIGURE 2. PILOT HOOD EMPLOYED FOR SIMULATED IFR FLYING
(VIEW FROM LEFT FRONT SEAT).

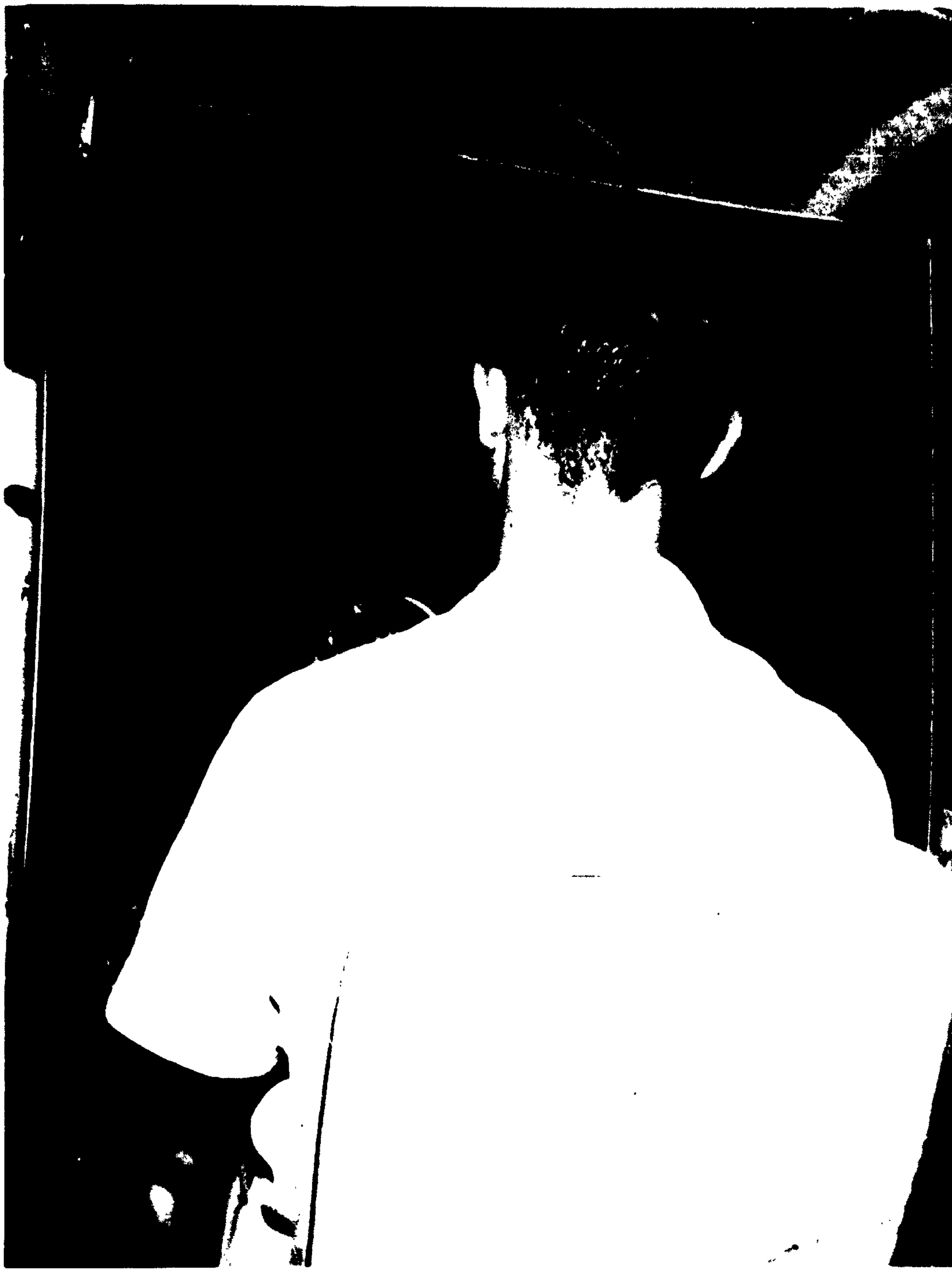


FIGURE 3. PILOT HOOD EMPLOYED FOR SIMULATED IFR FLYING (VIEW FROM THE REAR).

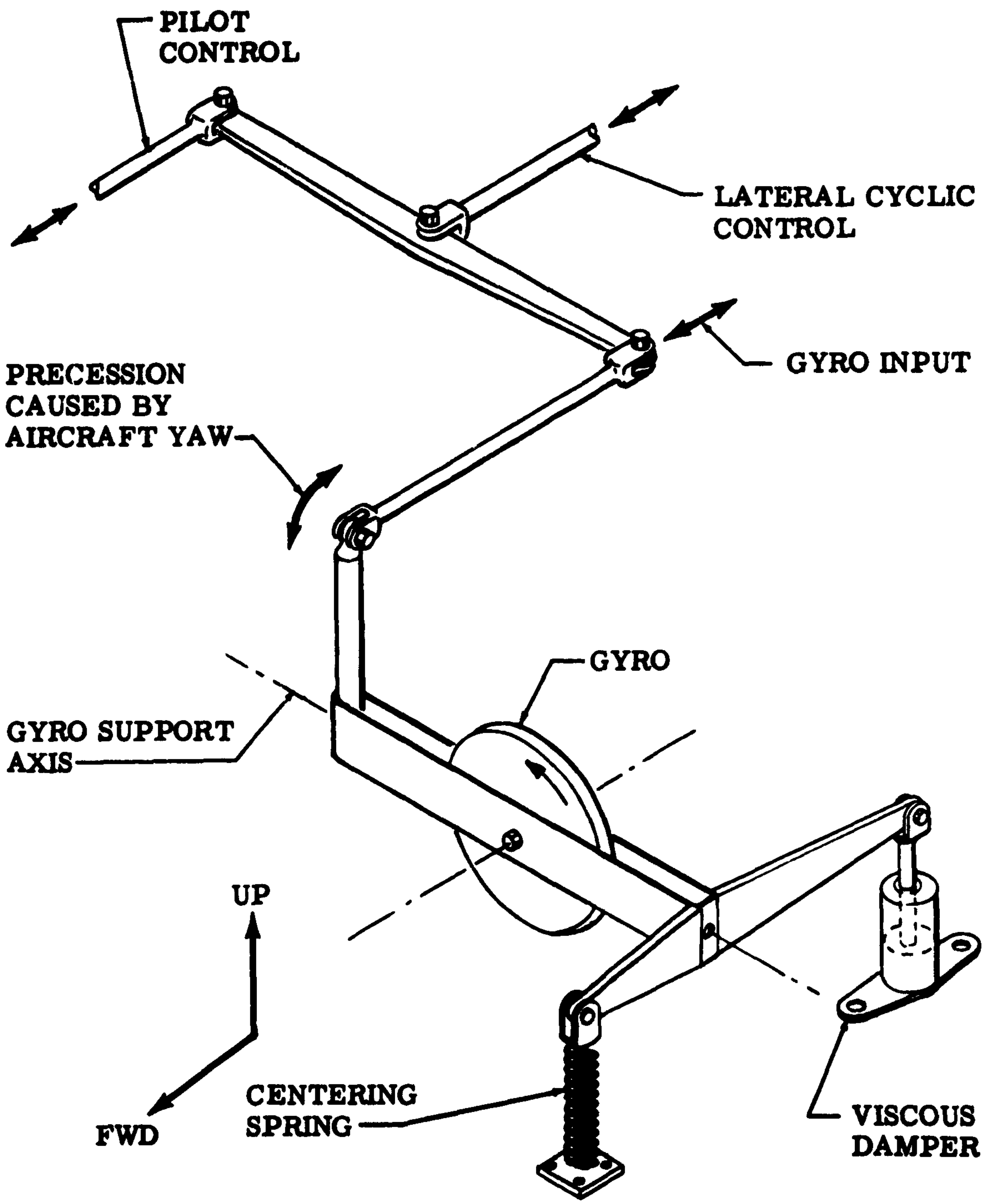
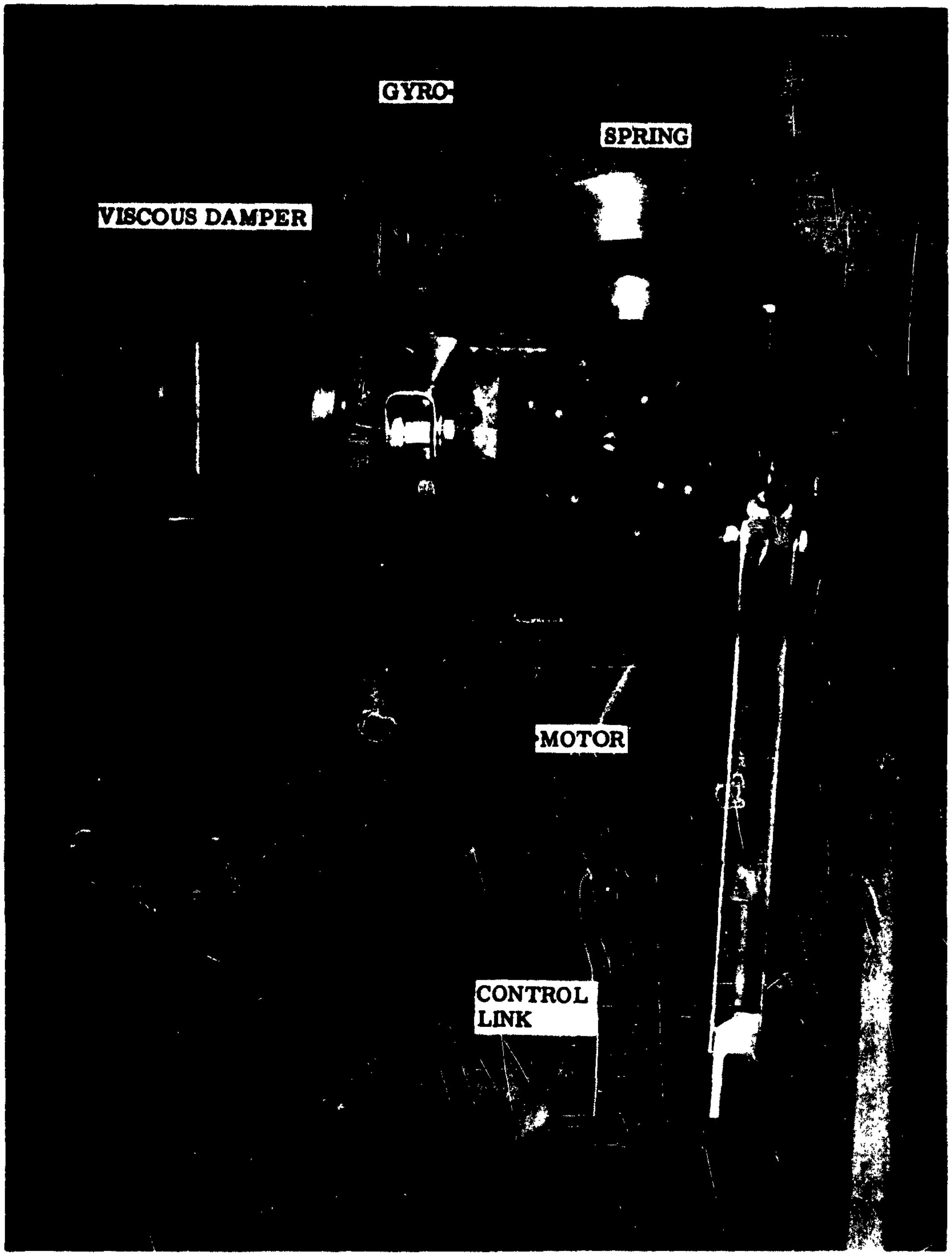


FIGURE 4. HEADING ASSIST GYRO SCHEMATIC.



Located in Front of the Left Front Seat. Provides Roll Correction for Heading Deviation

FIGURE 5. HEADING ASSIST GYRO INSTALLATION.

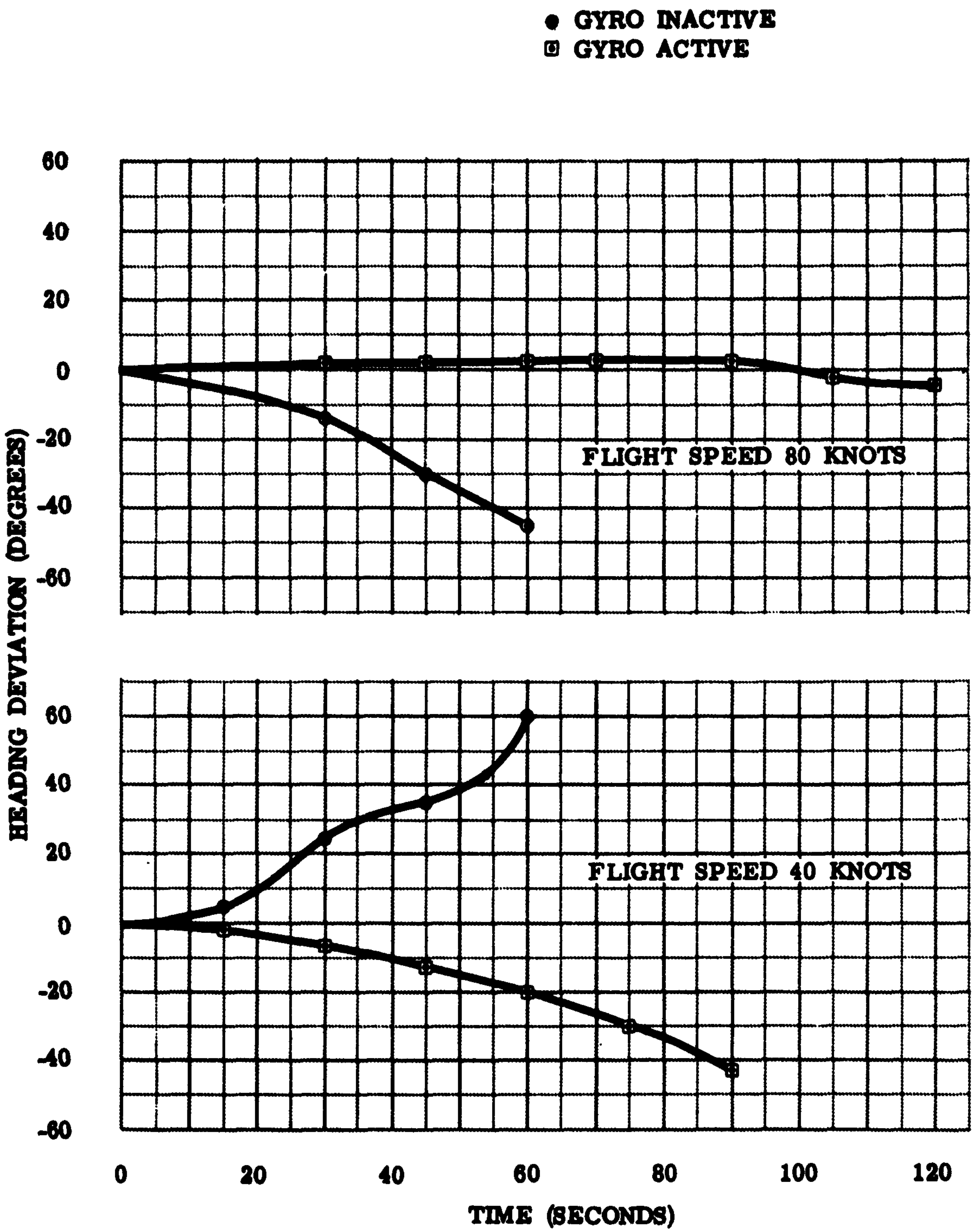
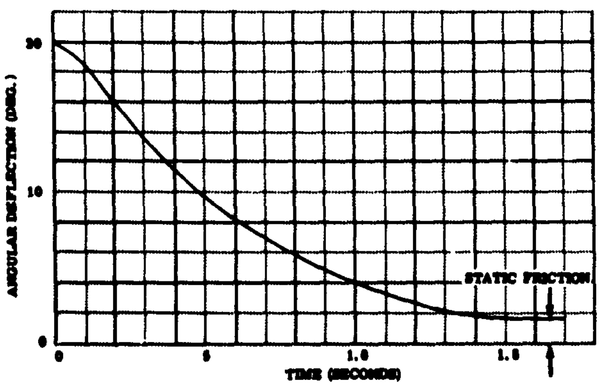


FIGURE 6. HEADING ASSIST GYRO PERFORMANCE WITH CONTROLS UNATTENDED.

HEADING ASSIST GYRO DATA SHEET

The heading assist gyro developed and tested on this program has the following basic parameters:

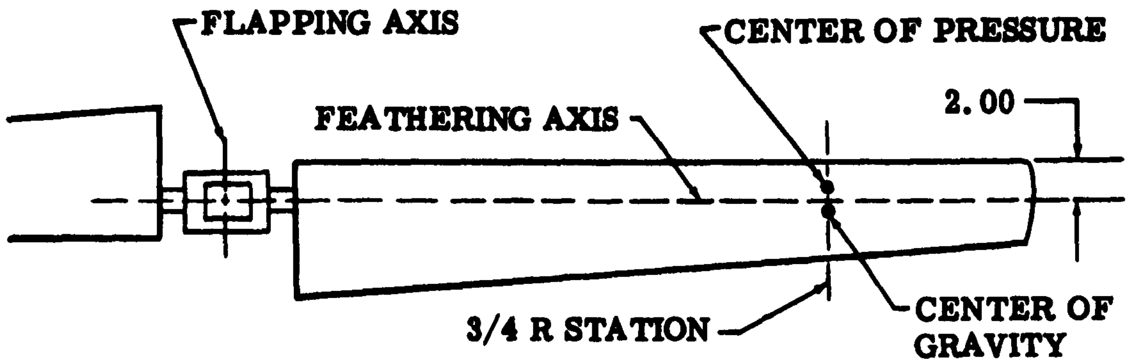
Gyro Authority	2.5% of the total lateral cyclic travel.
Maximum Gyro Roll Acceleration Capacity	$3.5°/sec^2$
Gyro Precessional Moment	$\frac{9.25 \text{ ft.\#}}{\text{rad./sec.}}$
Control System Static Friction	.15 ft.# (measured at gyro)
Gyro Angular Travel	± 20°
Gyro Centering Spring Constant	1.12 ft.#/rad.
Yaw Rate for Maximum Control	.042 rad./sec.
Yaw Rate at Threshold of Precession	.016 rad./sec.



The damper characteristics were determined to be nonlinear. The damper characteristics are defined below by the aperiodic curve. This curve presents the gyro motion resulting from release from a fully deflected position.

FIGURE 7. HEADING ASSIST GYRO DATA SHEET.

TAIL ROTOR A

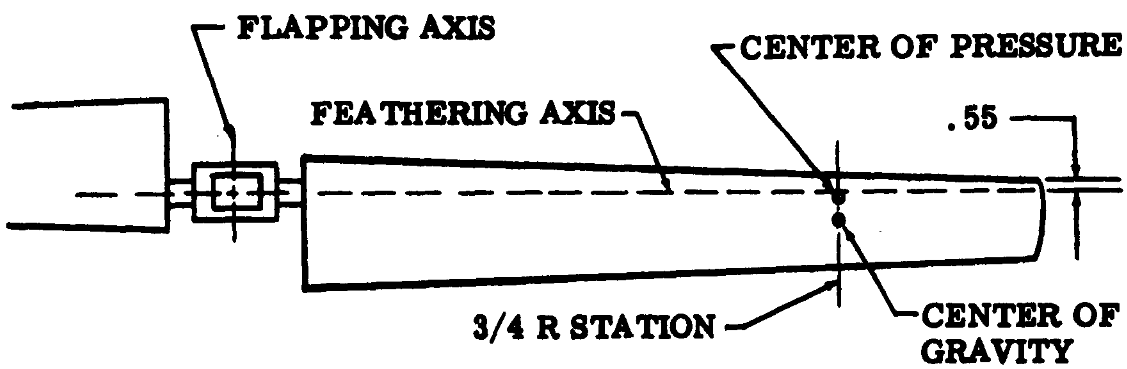


Semirigid Rotor

Center of Pressure .75 Ahead of Feathering Axis at 3/4 R

Center of Gravity .48 Behind Feathering Axis at 3/4 R

TAIL ROTOR B



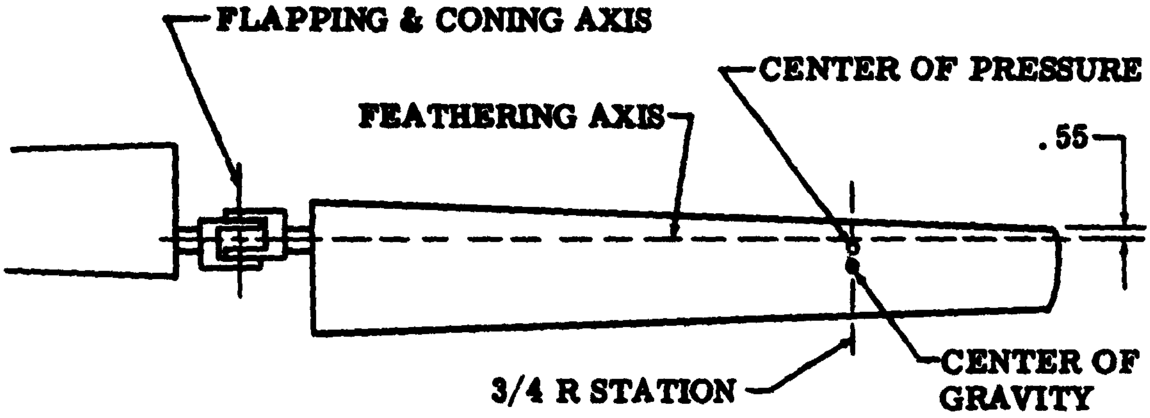
Semirigid Rotor

Center of Pressure .30 Behind the Feathering Axis at 3/4 R

Center of Gravity 1.50 Behind the Feathering Axis at 3/4 R

FIGURE 8. CONFIGURATION OF TAIL ROTORS A AND B.

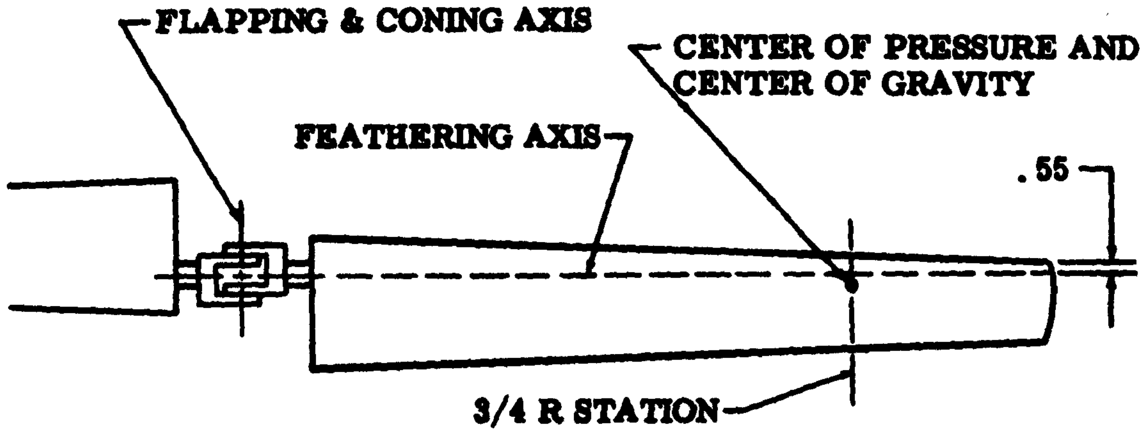
TAIL ROTOR C



Blades Individually Hinged to Flap and Cone
Center of Pressure .30 Behind the Feathering Axis at 3/4 R

Center of Gravity .81 Behind the Feathering Axis at 3/4 R

TAIL ROTOR D



Blades Individually Hinged to Flap and Cone
Center of Pressure and Center of Gravity .30 Behind Feathering Axis at 3/4 R.

FIGURE 9. CONFIGURATION OF TAIL ROTORS C AND D.

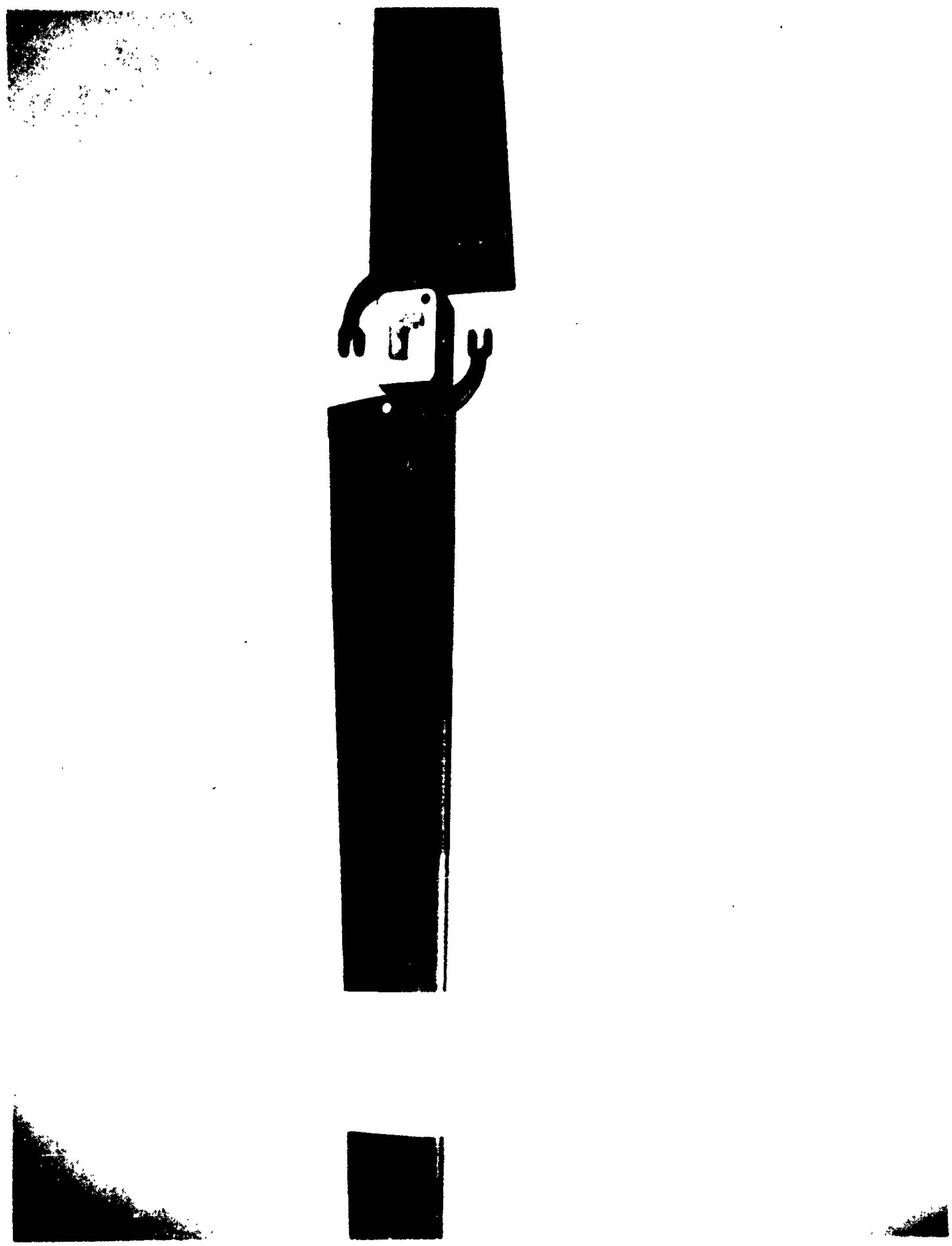


FIGURE 10. TAIL ROTOR A.

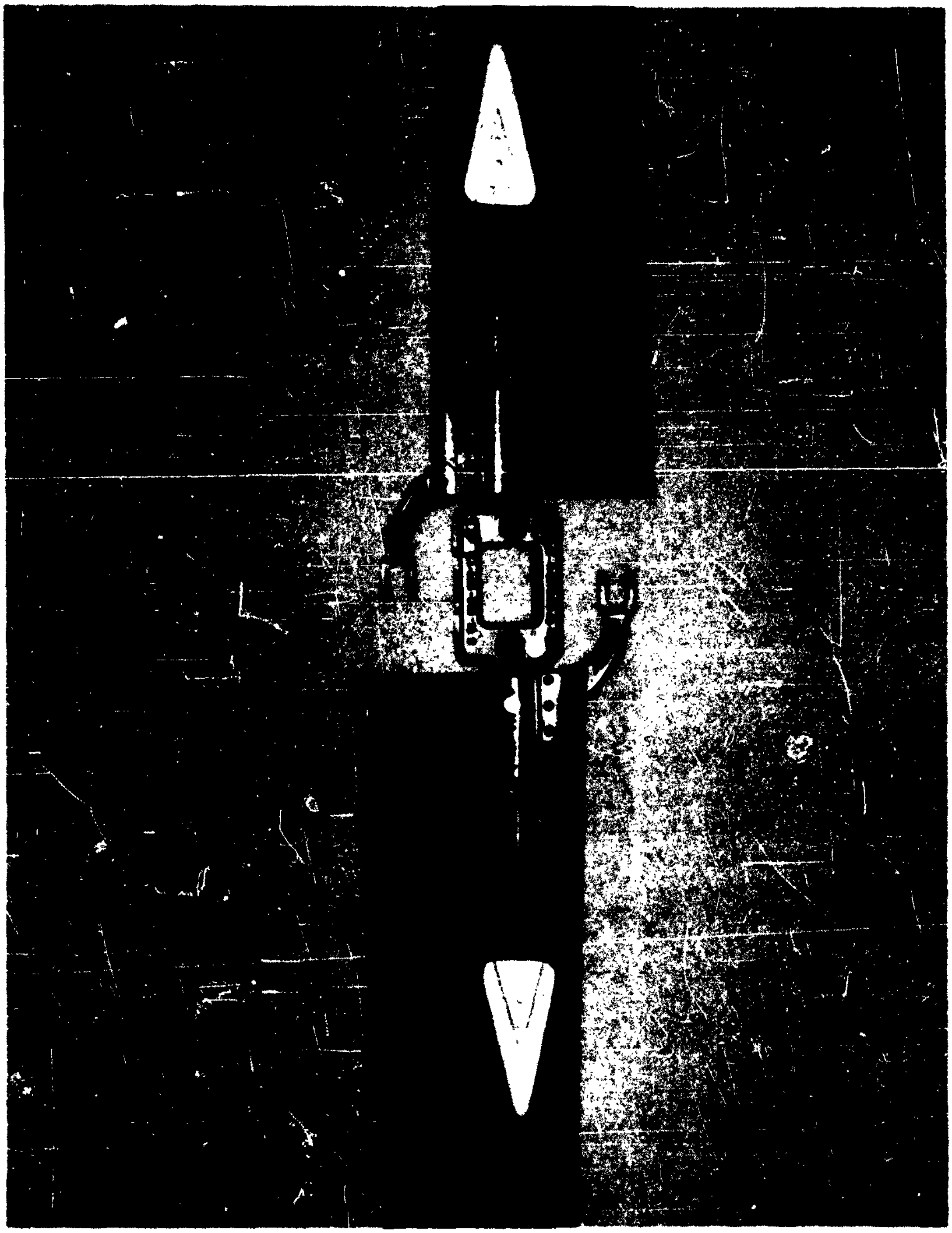


FIGURE 11. TAIL ROTOR B.

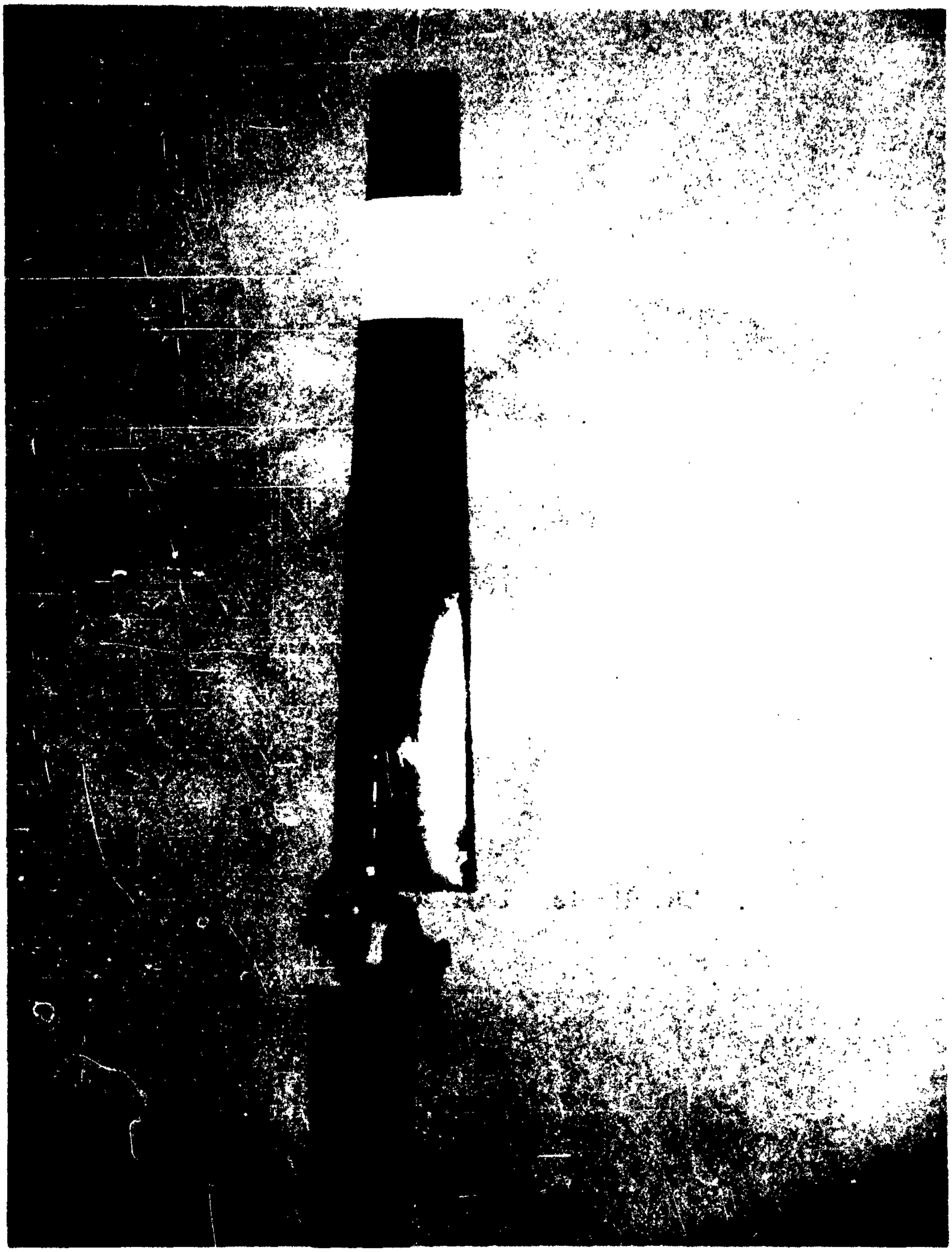


FIGURE 12. TAIL ROTOR C.

FIGURE 13. TAIL ROTOR D.

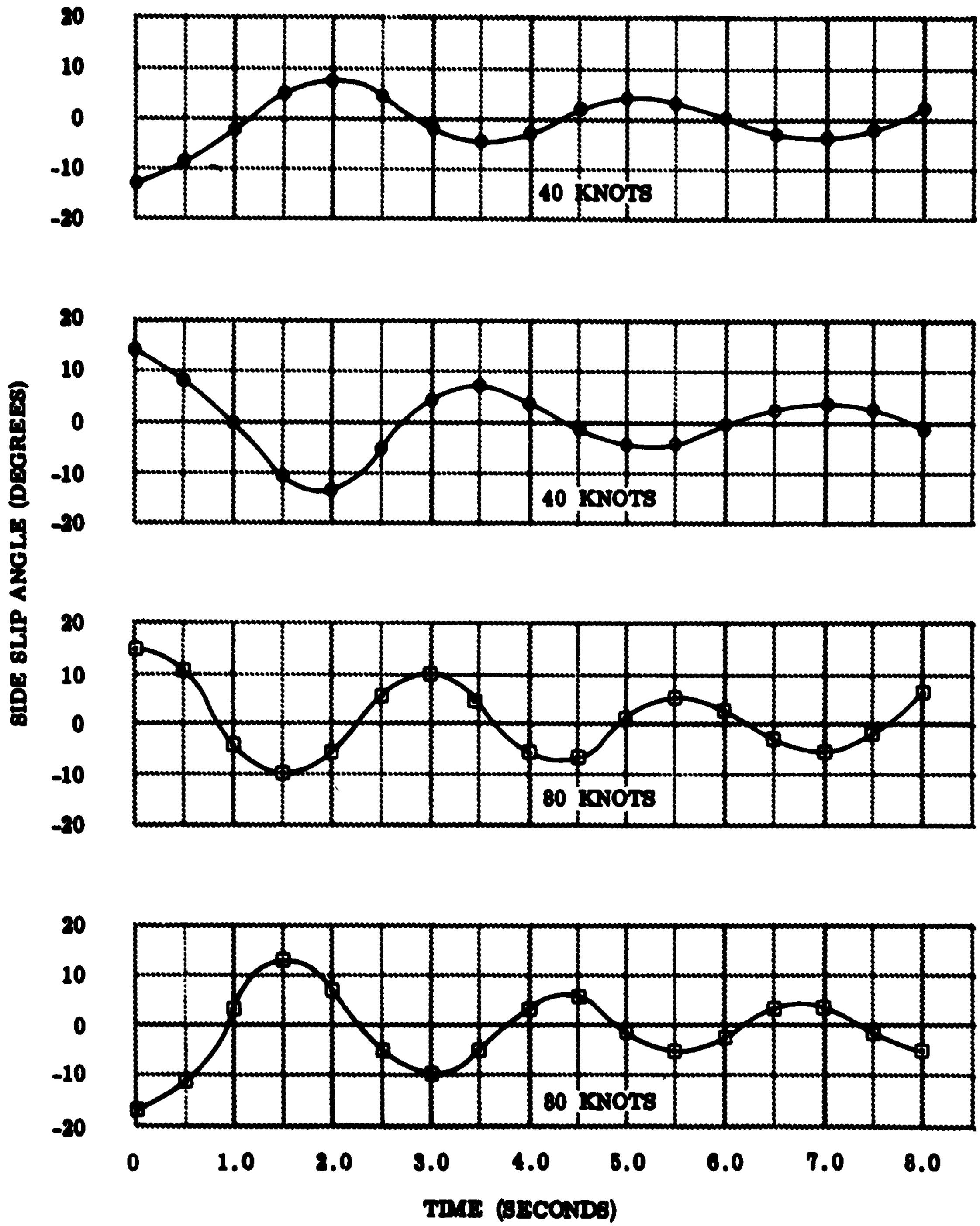


FIGURE 14. TAIL ROTOR A . HELICOPTER DIRECTIONAL REACTION FOLLOWING PEDAL PULSE.

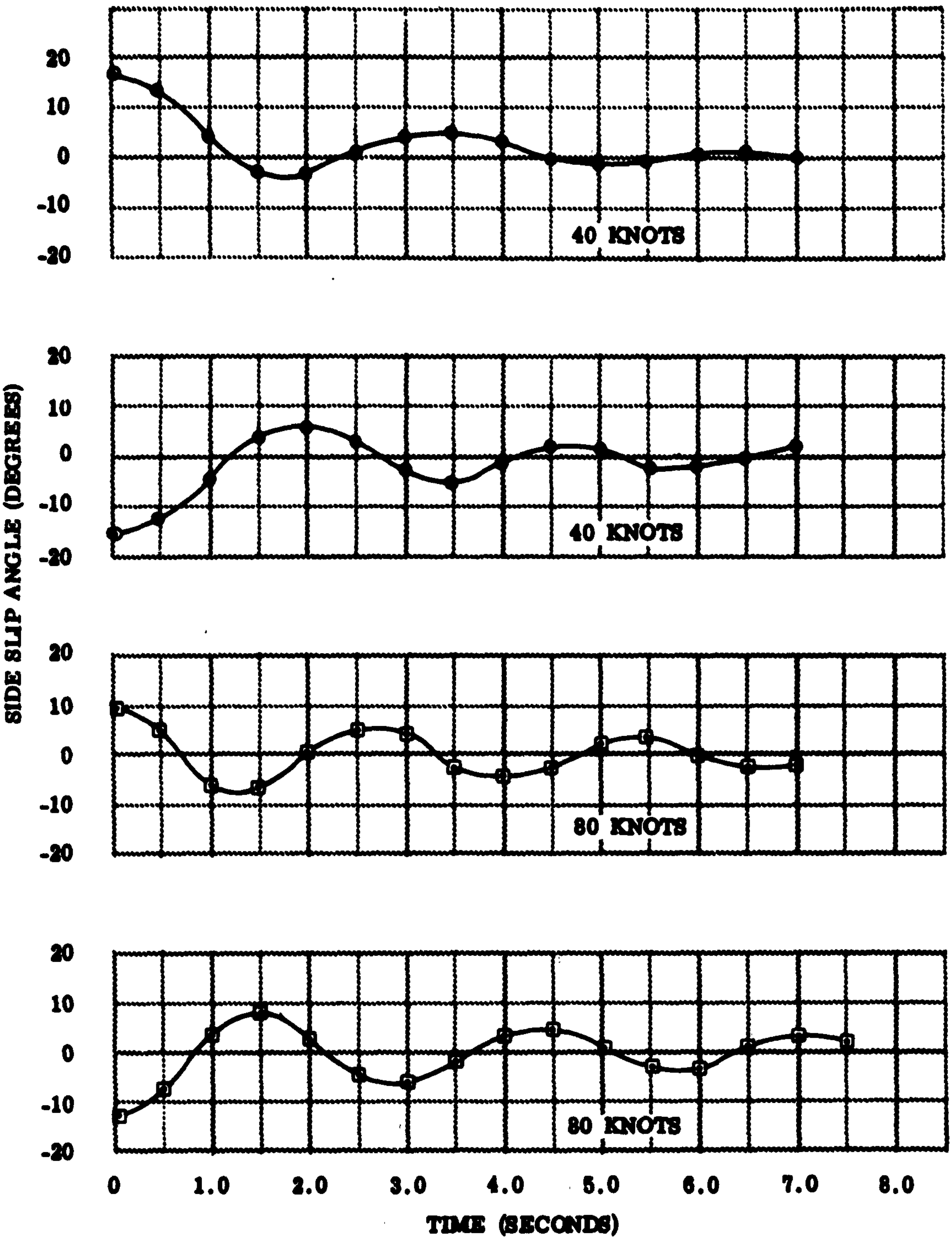


FIGURE 15. TAIL ROTOR B . HELICOPTER DIRECTIONAL REACTION FOLLOWING PEDAL PULSE.

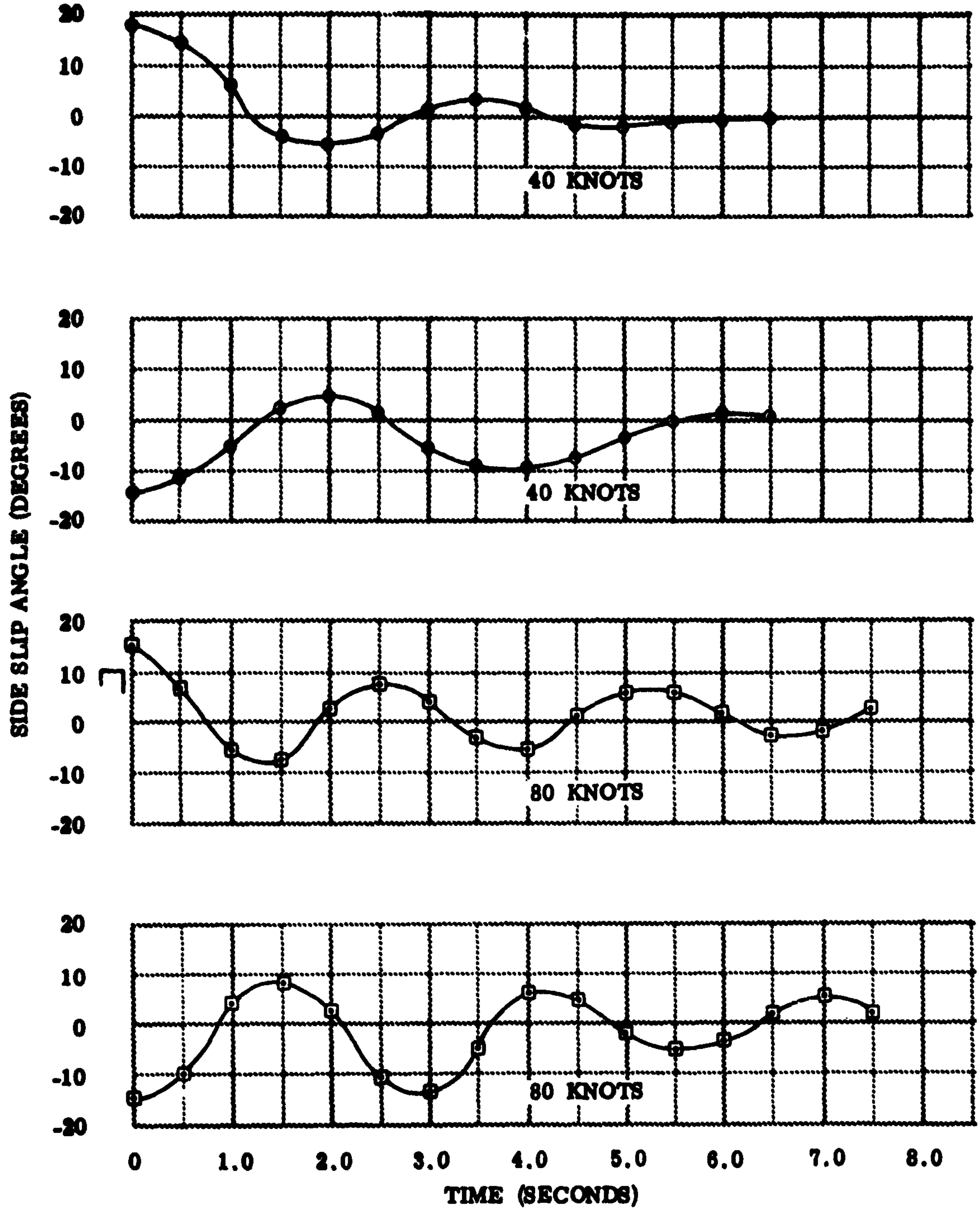


FIGURE 16. TAIL ROTOR C . HELICOPTER DIRECTIONAL REACTION FOLLOWING PEDAL PULSE.

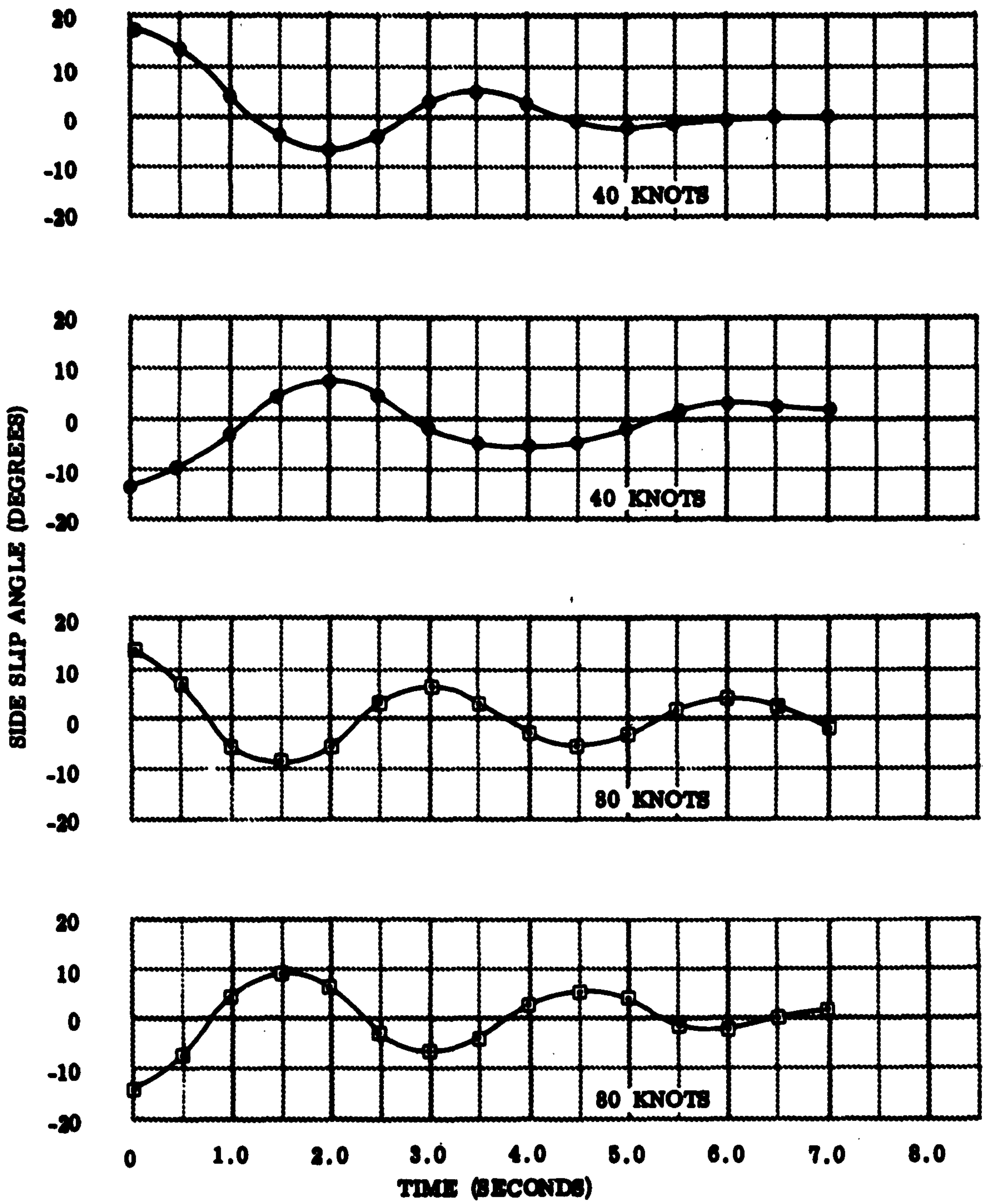


FIGURE 17. TAIL ROTOR D. HELICOPTER DIRECTIONAL REACTION FOLLOWING PEDAL PULSE.

Rotor Designation	A B	C	D
Blade Center of Gravity	48% Chord	35% Chord	25% Chord
Blade Coning Moment of Inertia	.21 Slug $Ft.^2$ (ABC)	.26 Slug $Ft.^2$	.26 Slug $Ft.^2$
Rotor Speed	1635 rpm		
Airfoil Section	NASA 0012		
Center of Pressure Location	25% Chord		
Chord @ 75% Radius	5.00 inches		
Blade Taper Ratio			
Control System Free Play	.5 Degree Blade Pitch Change		
Control System Deflection	17.5 Foot Pounds/Degree Blade Pitch Change		
Tail Rotor Thrust	17.4 Pounds/Degree Pitch Change		
Tail Rotor Moment Arm	21.5 Feet		
Yaw Axis Moment of Inertia	2170 Slug $Feet^2$		

FIGURE 18. TAIL ROTOR DATA SHEET.

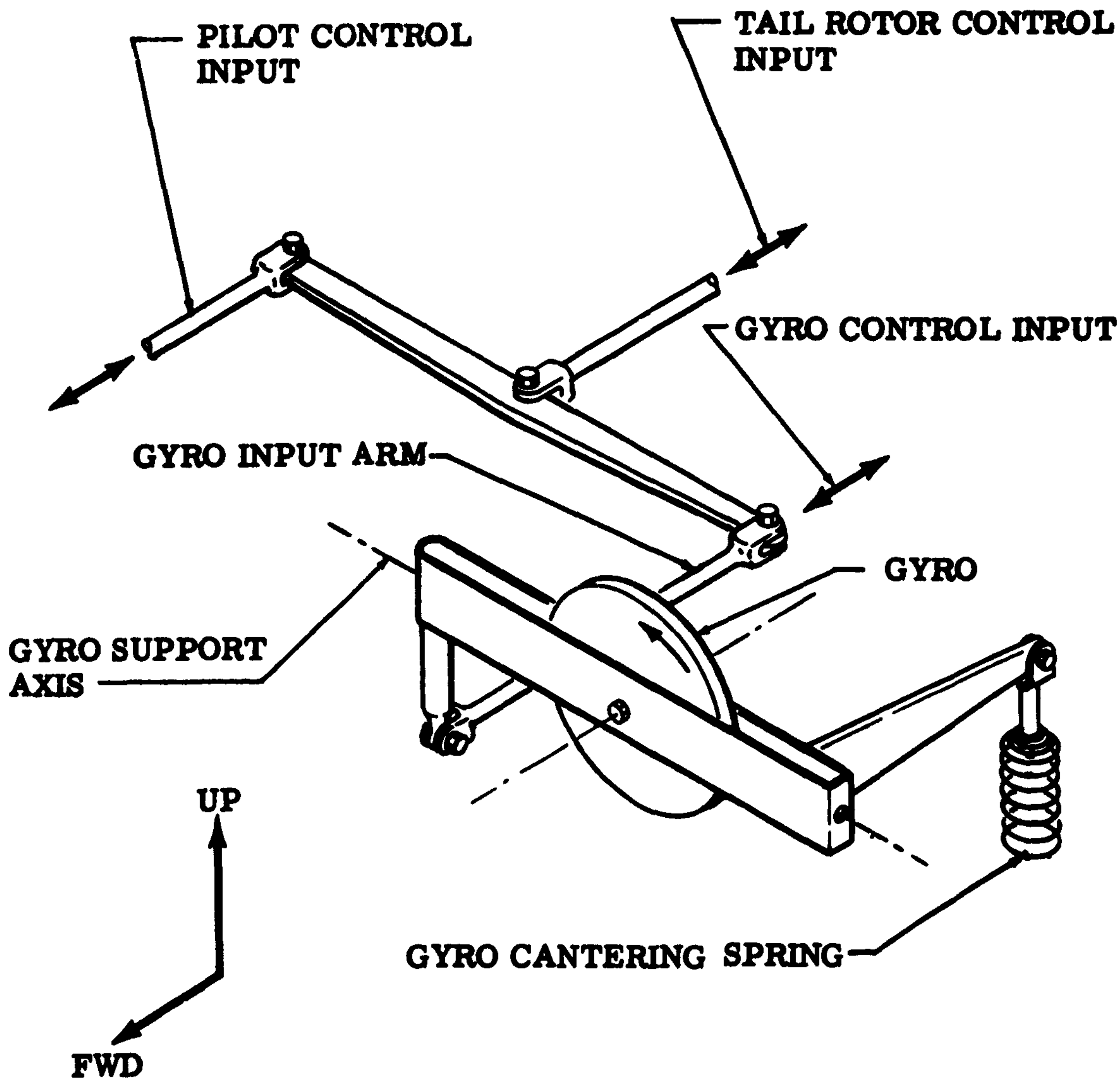
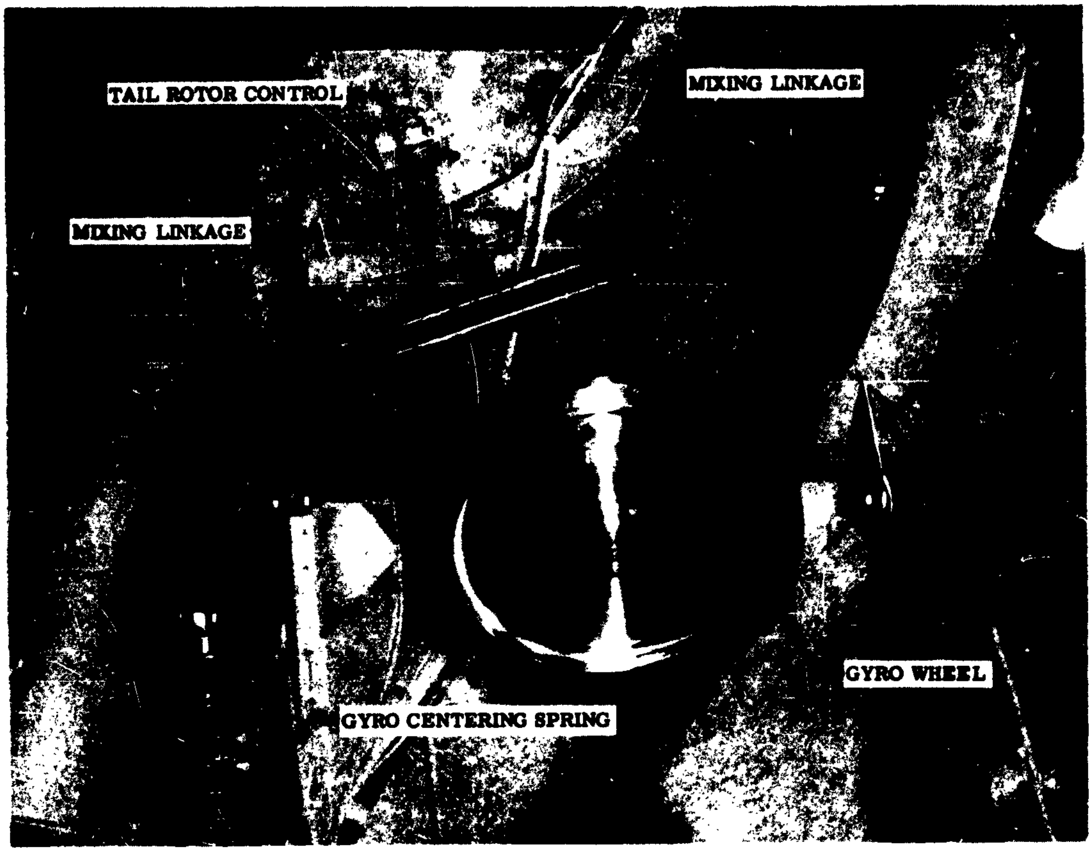
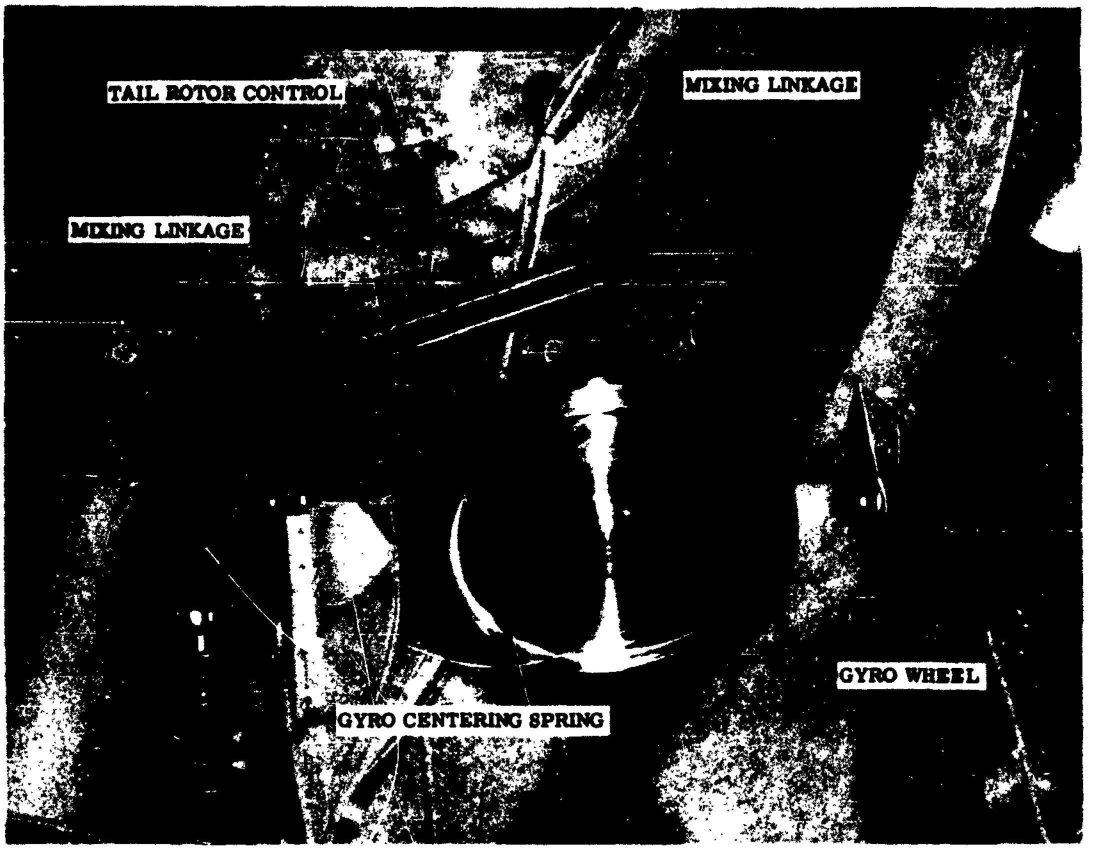


FIGURE 19. DIRECTIONAL DAMPING GYRO SCHEMATIC.



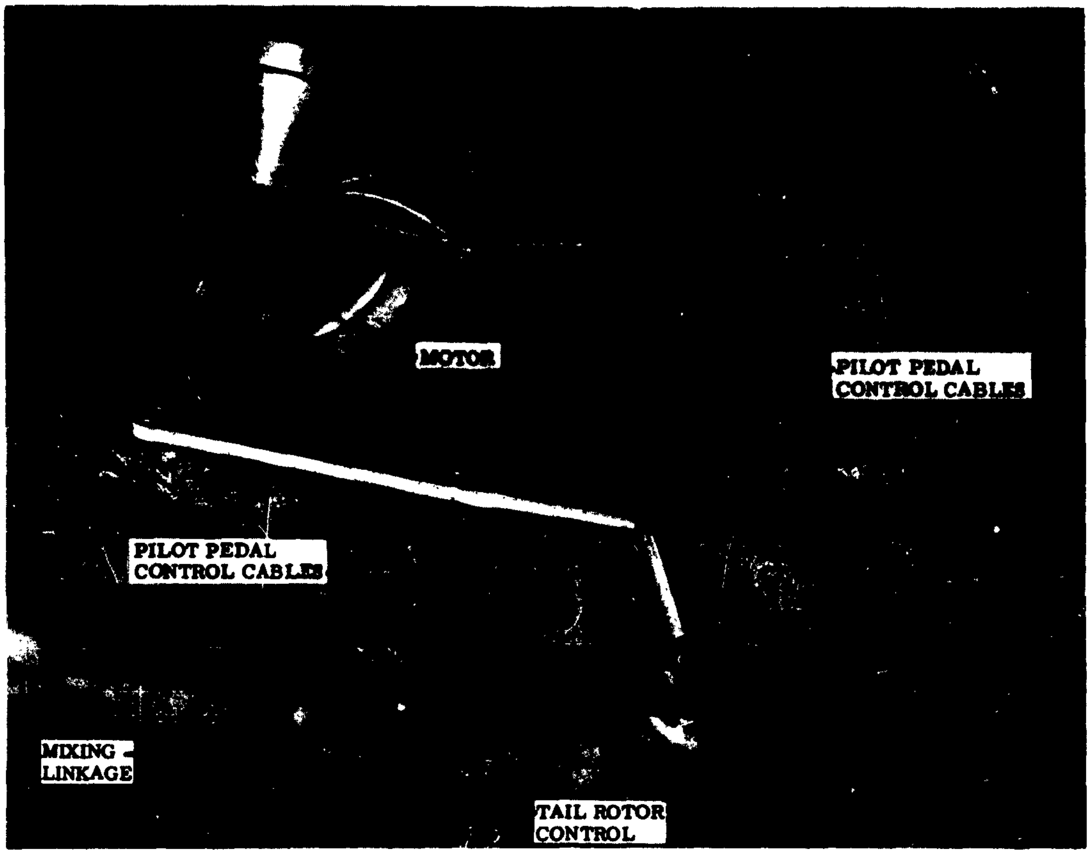
Senses Heading Deviation Rate and Provides Directional Correction.

FIGURE 20. DIRECTIONAL DAMPING GYRO (LOOKING REARWARD).



Senses Heading Deviation Rate and Provides Directional Correction.

FIGURE 20. DIRECTIONAL DAMPING GYRO (LOOKING REARWARD).

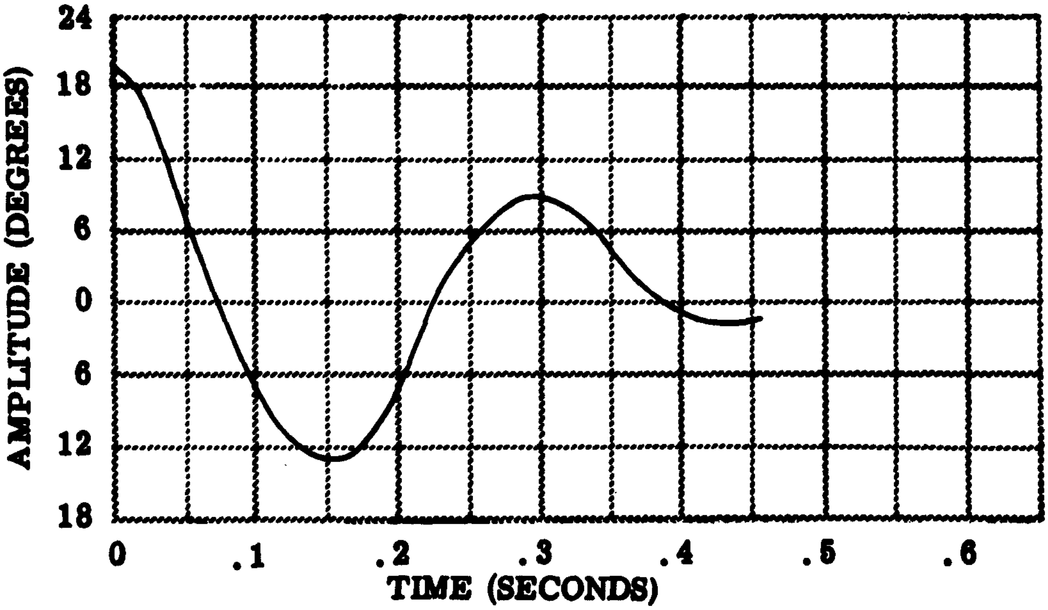


Located in the Forward Tail Boom.

FIGURE 21. DIRECTIONAL DAMPING GYRO (LOOKING FORWARD).

DATA SUMMARY SHEET
DIRECTIONAL DAMPING SYSTEM

Aircraft moment of inertia about yaw axis	=	2170 slug ft.^{2}
Maximum authority of damping system	=	$\pm$ 710 ft# = $\pm$ 7 1/2% of total control
Control system friction	=	.50 ft#
Control aerodynamic load variation	=	0 to .75 ft#
Gyro precessional moment	=	11.7 ft#/rad/sec
Gyro precession limits	=	$\pm$ 20°
Gyro centering spring constant	=	9.5 ft#/rad
Yaw rate of maximum gyro precession	=	.285 rad/sec
Yaw rate at threshold of gyro precession	=	.042 rad/sec



TIME HISTORY OF DAMPED GYRO MOTION

FIGURE 22. DIRECTIONAL DAMPING GYRO - DATA SHEET.

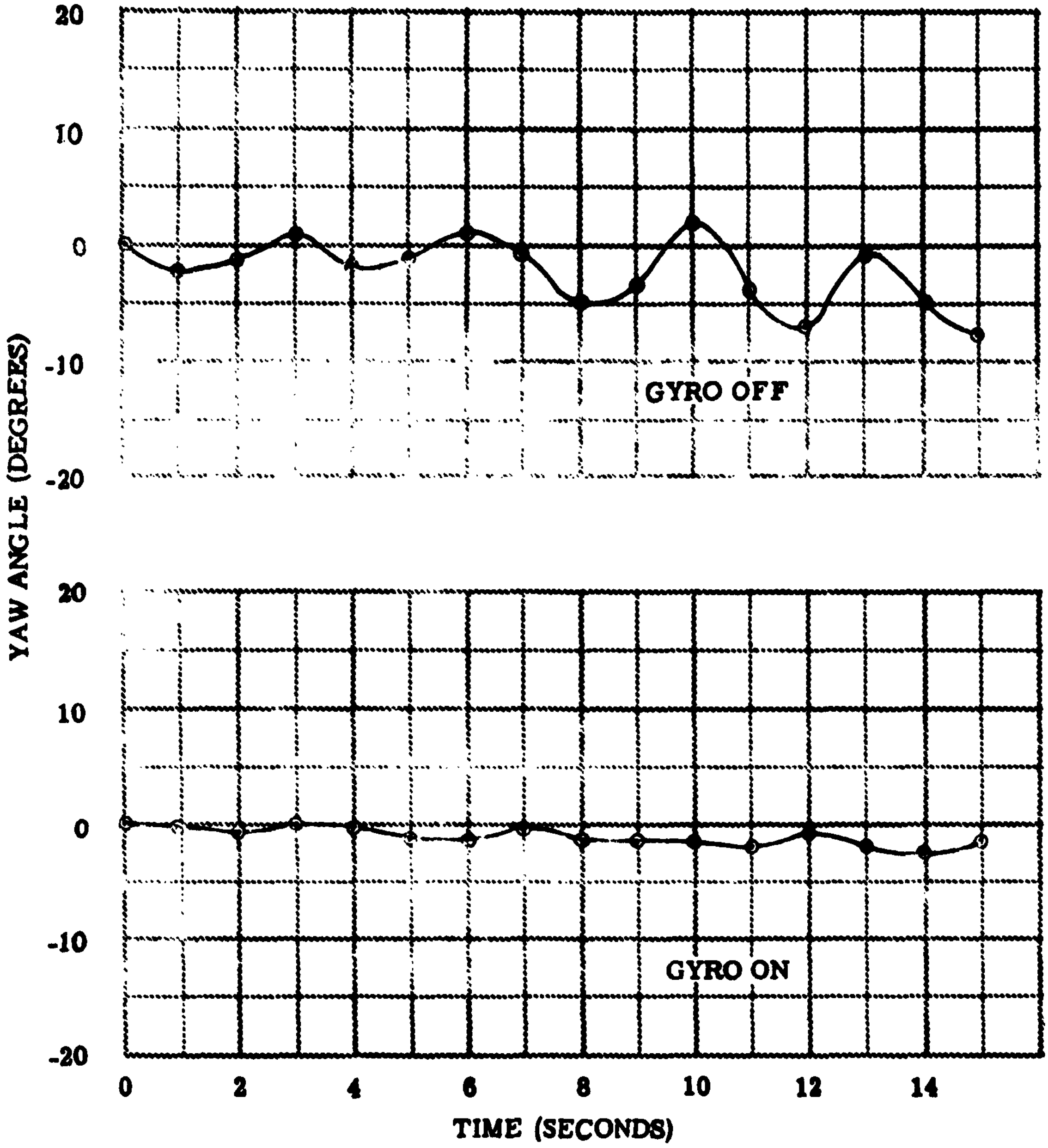


FIGURE 23. DIRECTIONAL DAMPING GYRO PERFORMANCE, UNATTENDED FLIGHT. AIRSPEED 80 KNOTS, LIGHT TURBULENCE.

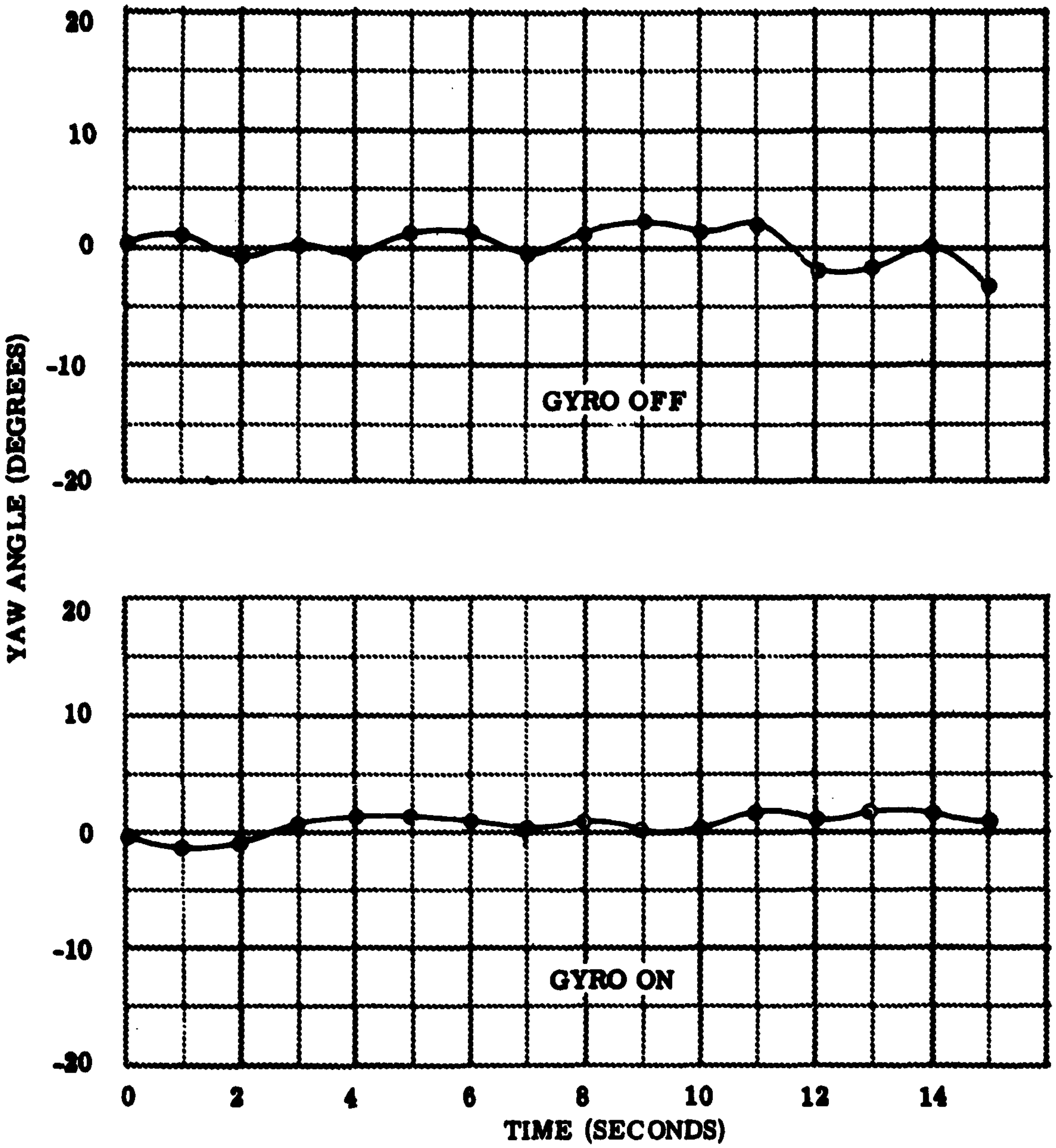


FIGURE 24. DIRECTIONAL DAMPING GYRO PERFORMANCE, UNATTENDED FLIGHT. AIRSPEED 80 KNOTS, LIGHT TURBULENCE.

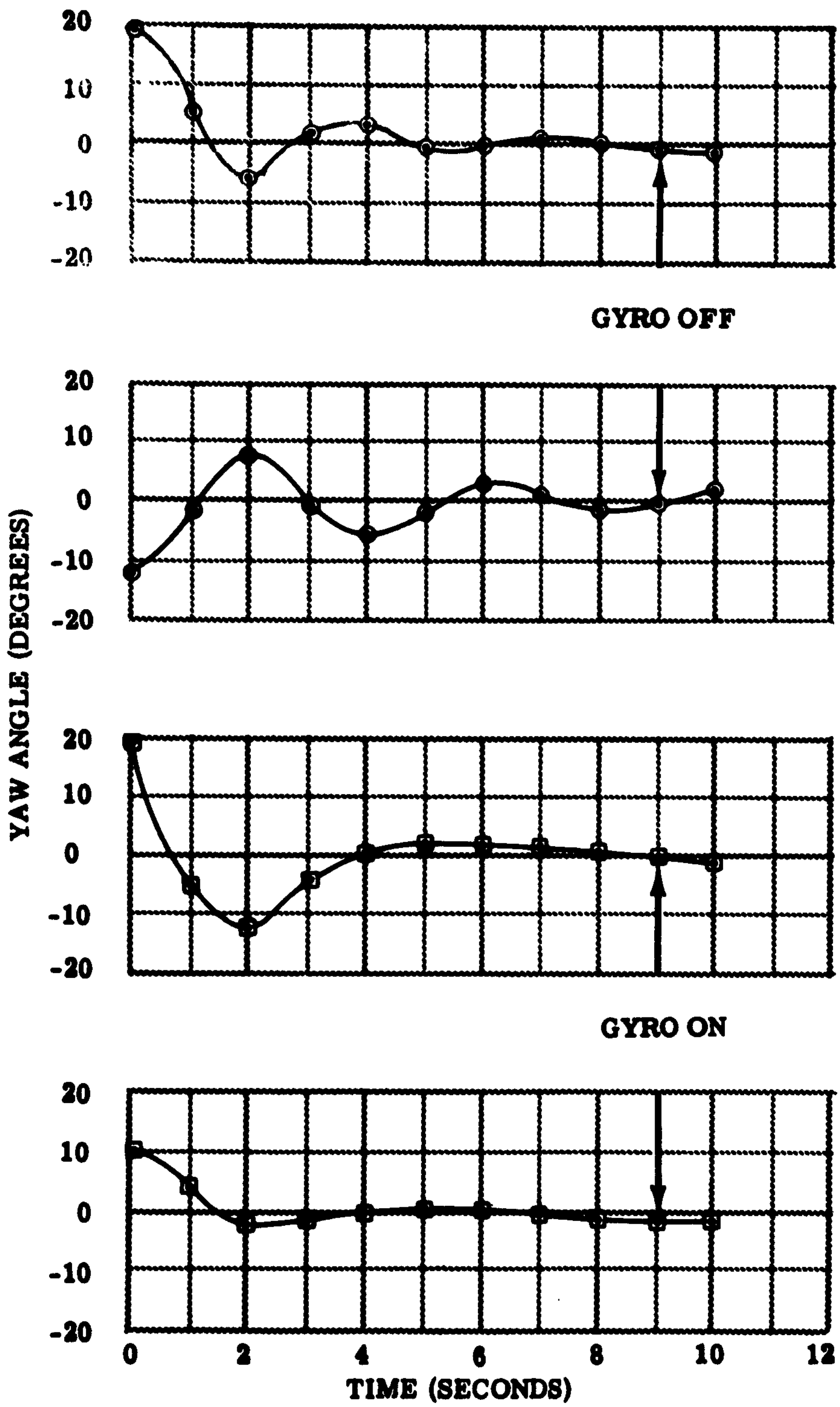


FIGURE 25. DIRECTIONAL DAMPING GYRO PERFORMANCE, FOLLOWING PEDAL PULSE. AIRSPEED 40 KNOTS.

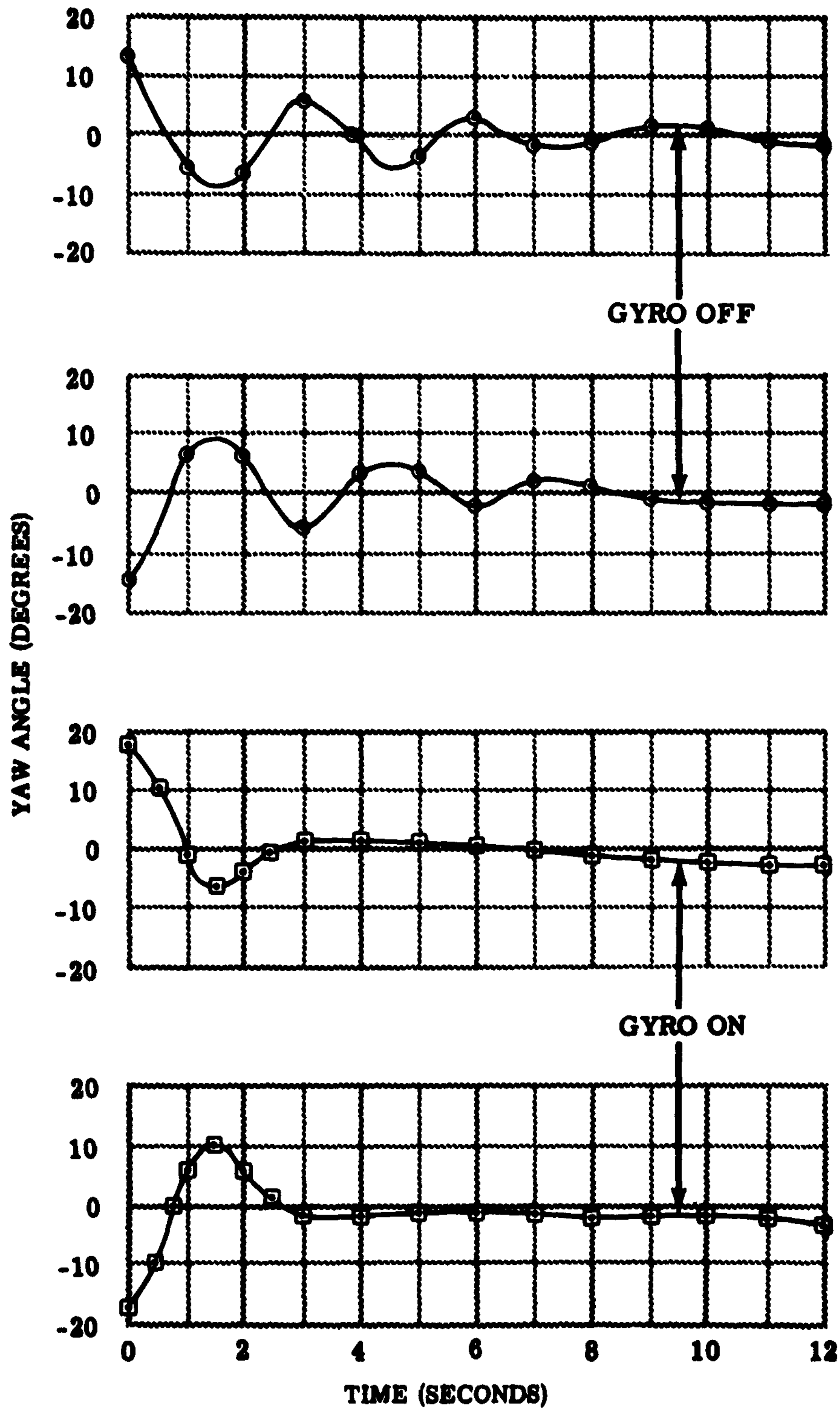
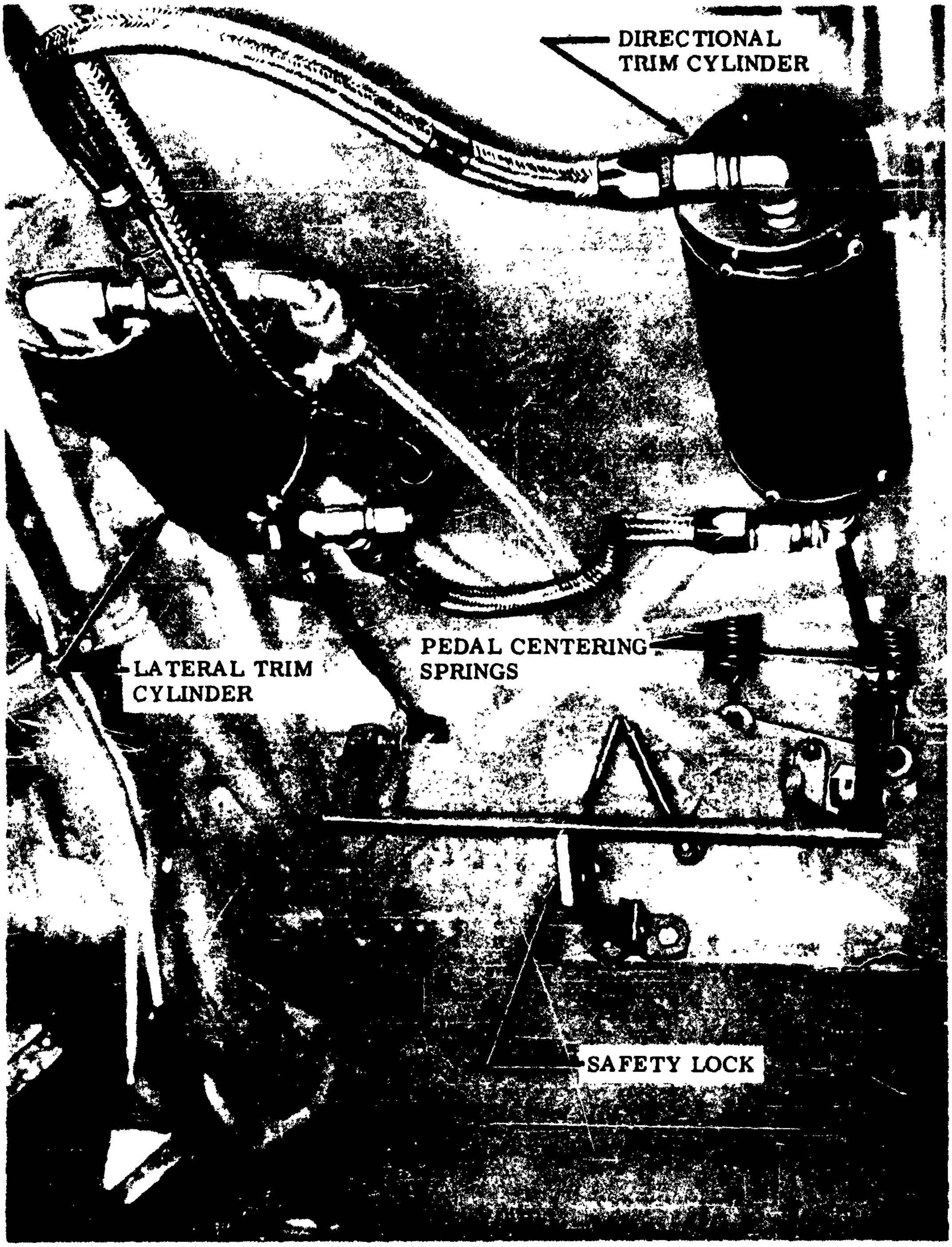


FIGURE 26. DIRECTIONAL DAMPING GYRO PERFORMANCE, FOLLOWING PEDAL PULSE. AIRSPEED 80 KNOTS.



Located Below Engine. Senses Power and Provides Directional and Lateral Trim Corrections.

FIGURE 27. AUTOMATIC TRIM UNITS.

1856-63

DISTRIBUTION

USCONARC	3
First US Army	3
Second US Army	2
Third US Army	2
Fourth US Army	1
Sixth US Army	1
USAIC	2
USACGSC	1
USAWC	1
USAATBD	1
USAARMBD	1
USAAVNBD	1
USATMC(FTZAT), ATO	1
USAPRDC	1
DCSLOG	2
Rsch Anal Corp	1
ARO, Durham	2
OCRD, DA	1
USATMC Nav Coord Ofc	1
NATC	2
CRD, Earth Scn Div	1
USAAVNS, CDO	1
DCSOPS	1
OrdBd	1
USAQMCDA	1
QMFSA	1
USACECDA	1
USATCDA	1
USATB	1
USATMC	20
USATC&FE	4
USATSCH	3
USATRECOM	17
USATTCA	1
USA Tri-Ser Proj Off	1
TCLO, USAABELCTBD	1
USASRDL LO, USCONARC	2
USATTCP	1
OUSARMA	1
USATRECOM LO, USARDG (EUR)	1

USAEWES	1
TCLO, USAAVNS	1
USATDS	5
USARPAC	1
EUSA	1
USARYIS/IX CORPS	2
USATAJ	6
USARHAW	3
ALFSEE	2
USACOMZEUR	3
USARCARIB	4
AFSC (SCS-3)	1
APGC(PGAPI)	1
Air Univ Lib	1
AFSC (Aero Sys Div)	2
ASD (ASRMPT)	1
CNO	1
ONR	3
BUWEPS, DN	5
ACRD(OW), DN	1
BUY&D, DN	1
USNPGSCH	1
Dav Tay Mod Bas	1
CMC	1
MCLFDC	1
MCEC	1
MCLO, USATSCH	1
USCG	1
USASGCA	1
Canadian LO, USATSCH	3
BRAS, DAQMG(Mov & Tn)	4
USASG, UK	1
NAFEC	3
Langley Rsch Cen, NASA	2
Geo C. Marshall Sp Flt Cen, NASA	1
MSC, NASA	1
Ames Rsch Cen, NASA	2
Lewis Rsch Cen, NASA	1
Sci & Tech Info Fac	1
USGPO	1
ASTIA	10
HumRRO	2

US Patent Ofc, Scn Lib	1
USAMOCOM	3
USAMC	7
USSTRICOM	1
ASD, FCL	1
Cessna Acft Co	10

Cessna Aircraft Company, Wichita, Kansas. AN INVESTIGATION OF MECHANICAL STABILITY AND TRIM AUGMENTATION ON HELICOPTER IFR CAPABILITY - Charles M. Seibel, TRECOM Technical Report 63-18, June 1963, 48 pp. (Contract DA 44-177-TC-791) USATRECOM Task 1D121401A14174 (Formerly Task 9R38-01-017-74).

Unclassified Report
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1. Mechanical Stability & Trim Augmentation on Helicopter IFR Capability
2. Contract DA 44-177-TC-791

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